Comparative analysis of new configurations of aircraft aimed at competitiveness, environmental compatibility and safety

Doctoral Dissertation
Doctoral Program in Aerospace Engineering (29th Cycle)

Comparative analysis of new configuration of aircraft aimed at competitiveness, environmental compatibility and safety

By

Roberta Fusaro

******

Supervisor(s):
Prof. Nicole Viola, Supervisor
Prof. Sabrina Corpino, Co-Supervisor

Doctoral Examination Committee:
Prof. Tomasz Goetzendorf-Grabowski, Warsaw University of Technology
Prof. Andreas Strohmayer, University of Stuttgart
Dr. Martin Sippel, Deutschen Zentrums für Luft- und Raumfahrt (DLR) - Bremen
Prof. Eugenio Brusa, Politecnico di Torino
Prof. Paolo Maggiore, Politecnico di Torino

Politecnico di Torino
2017

Declaration

I hereby declare that, the contents and organization of this dissertation constitute my own original work and does not compromise in any way the rights of third parties, including those relating to the security of personal data.

Roberta Fusaro

2017

* This dissertation is presented in partial fulfillment of the requirements for **Ph.D. degree** in the Graduate School of Politecnico di Torino (ScuDo).

I would like to dedicate this thesis to all those people that allow my dreams to come true, and especially to Sergio, who passed away too early, but whose teaching and memory will stay forever in my hearth

Acknowledgment

And I would like to acknowledge the entire research group of Politecnico di Torino that support me during these years in carrying out the several research activities in which I had the pleasure to be involved with.

I would like to thank you my Supervisors, Nicole and Sabrina, who followed me all along this adventure giving to me all the technical and personal support to carry out this work.

And eventually, a special thank you to my parents and to Davide who strongly encouraged me never give up.

Abstract

This Ph.D. Thesis aims at suggesting a proper integrated and multidisciplinary design methodology to improve the current conceptual and preliminary design phases of breakthrough innovative aerospace products. The methodology, based on a Systems Engineering approach, is presented together with an envisaged tool-chain, consisting of both commercial and ad-hoc developed software, integrated in a Model-Based Systems Engineering perspective. In addition, for the sake of clarity and for validation purposes, a specific case study has been selected and developed all along the document. The reference case-study is inspired to a real pre-feasibility study in which the research group of Politecnico di Torino, which the author of this Thesis belongs to, has been involved. The project aims at developing a suborbital vehicle able to perform parabolic flights for both scientific and touristic purposes. This kind of initiatives paves the way for the future hypersonic vehicles, because it allows to crucial enabling technologies to be tested and validated in relevant environment but with lower performances' requirements.

The Thesis is articulated in seven Chapters with an introduction and conclusion sections and in each Chapter a balanced mix between theoretical investigation, mathematical model development, tool selection or development and application to the selected case study is guaranteed. This document starts reporting the major reasons why an innovative design methodology should be envisaged to deal with the increasing level of complexity in the aerospace domain.

In particular, in the first Chapter, a brief overview of existing or under-development initiatives related to hypersonic is reported, together with the description of the different types of mission in which the new hypersonic vehicles will be exploited. Moreover, the major issues related to the infrastructures required to operate these transportation systems are summarized. As far as operations are concerned, a short section makes the readers aware of the current under-development regulatory framework.

Then, the integrated multidisciplinary design methodology is presented starting from the very high level analyses up to the sizing of the different components of the transportation system. All along the document, crucial role is played by requirements, whose management can allow a complete traceability of the different design characteristics during the overall product life-cycle. Furthermore, proper algorithms allowing to move from purely qualitative to quantitative trade-offs, are presented, with a noticeable advantage in terms of traceability and reproducibility.

Eventually, further improvements of both the tool-chain and the reference case studies are envisaged for future developments.

Contents

Chapter 2

Chapter 3

Chapter 4

Chapter 5

Chapter 6

Chapter 7

List of Figures

List of Tables

Acronyms

A	
ALTEC	Aerospace Logistics Technology Engineering Company
AD	Activity Diagram
AD^2	Advancement Degree of Difficulty
AHP	Analytical Hierarchy Process
ARD	Atmospheric Re-entry Demonstrator
ARV	Ascent and Re-entry Vehicle
ASI	Italian Space Agency
ATR	Air Turbo Ramjet
AVATAR	Aerobic Vehicle for Advanced Trans-Atmospheric Research

B	
BDD	Block Definition Diagram
BGRV	Boost Glide Re-entry Vehicles

C	
CAA	Civil Aviation Authority
CAV	Cruise and Acceleration Vehicle
CCM	Change and Configuration Management
CG	Centre of Gravity
ConOps	Concept of Operations

D	
DMR	Dual Mode Ramjet

E	
EASA	European Aviation Safety Agency
EC	European Community
ECLSS	Environmental Control and Life Support Subsystem
ESA	European Space Agency
EU	**European Union**

F	
FAI	International Aeronautical Federation
FFBD	Functional Flow Block Diagram
FoM	Figure of Merit

G	
GTO	Geostationary Transfer Orbit

H	
HOTOL	HOrizontal Take Off and Landing
HRE	Hypersonic Research Engine

I	

IBD	Internal Block Diagram
ICAO	International Civil Aviation Organization
ICBM	InterContinental Ballistic Missile
IRL	Integration Readiness Level
IXV	Intermediate eXperimental Vehicle
J	
K	
L	
LACE	Liquid Air Cycle Engine
LAPCAT	Long-Term Advanced Propulsion Concepts and Technologies
L/D	Lift/Drag ratio
LEO	Low Earth Orbit
M	
MA	Mission Analysis
MBSE	Model Based Systems Engineering
MoU	Memorandum of Understanding
MTOM	Maximum Take-Off Mass
N	
NACA	National Advisory Committee for Aeronautics
NASA	National Aeronautics and Space Administration
NASP	National Aero-Space Plane
NMSA	New Mexico Spaceport Authority
NW-RV	Non-Winged Re-entry Vehicle
O	
O-ARV	Orbital Ascent and Re-entry Vehicle
OEI	One Engine Inoperative
P	
P2P	Point to Point
Q	
QFD	Quality Functional Deployment Tools
R	
RAMS	Reliability Availability Maintainability and Safety
RBCC	Rocket Based Combined Cycle
RIVET	Reverse Installation Vectored Engine Thrust
RLV	Reusable Launch Vehicle
ROI	Return on Investments
RV	Re-entry Vehicles
S	
SABRE	Synergistic Air-Breathing Rocket Engine
SARPs	Standards and Recommendations Practices
S3	Swiss Space System
SE	Systems Engineering
SERJ	Supercharged Ejector Ramjet
SMD	State Machines Diagrams
SMV	Space Maneuver Vehicle
SO-ARV	SubOrbital Ascent and Re-entry Vehicle
SoS	System of Systems
SRL	System Readiness Level
SS1	SpaceShipOne
SSTO	Single Stage To Orbit
T	
TAS-I	Thales Alenia Space - Italy
TBCC	Turbine Based Combined Cycle
TCS	Thermal Control

	Subsystem
TPS	Thermal Protection Subsystem
TRL	Technology Readiness Level
TSTO	Two Stage To Orbit
U	
UCD	Use Case Diagram
UN	United Nations
V	
VTOL	Vertical Take-Off and Landing
W	
WBS	Work Breakdown Structure
W-RV	Winged Re-entry Vehicle
X	
Y	
Z	

Introduction

The need to go faster

Living in the second decade of the 21st century, fully submerged in and surrounded by high-tech devices and related services, it is evident that one of the main features of our time is the increasing rate of development of new technologies. This growing trend is absolutely positive, even if a unique speed or trend is not easily identifiable in this process and different discontinuities can be highlighted. Cyclical brakes could be identified and they are mainly due to adverse economic circumstances but they alternate with favourable historical periods, like the two World Wars, in which the rush for inventing, developing and exploiting new technologies was encouraged from different sides.

In particular, the aeronautical and the aerospace engineering domains are a crystal clear example of technological fields with a current increasing speed of technological development. In the last decades, the aeronautical and aerospace engineering fields have been affected by the so-called *phenomenon of convergence of interests*, highlighted in Figure 1 mainly in terms of altitude. This trend can be justified considering that aeronautics has a great interest in developing faster transportation systems, increasing the terrestrial net of connections and to reach this goal, flight altitude should increase, moving to those part of the atmosphere that were typically considered part of the space domain. On the other hand, the development of innovative technologies can allow space engineers to overcome problems facing when they are approaching these high atmospheric layers during re-entry. Indeed, the development of new materials, technologies and systems able to allow hypersonic flight, surviving at very harsh environments are considered with a great interest by space enterprises with the aim of exploiting them also in interplanetary missions or even more in entry or re-entry missions, not only on Earth but also on other planets or celestial bodies.

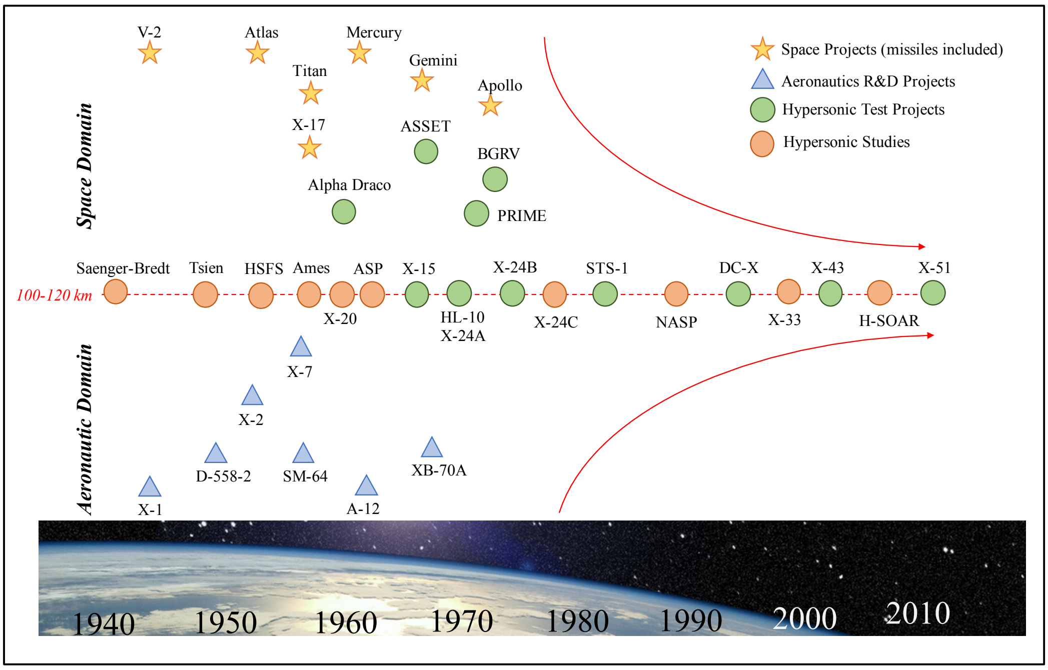


Figure 1: Air and space convergence

Thus, hypersonic flights can be considered the ring of conjunction between aeronautical engineering and space engineering and this tendency is re-designing the borders of both these domains. They are modifying their respective interests not only in terms of missions and services provided to the market, but also in terms of minimum and maximum altitude of direct competence and even more in terms of laws and regulations (Jakhu, 2011). Moreover, also from the technical point of view, researchers have to face with several problems related, for example, to the TRL (Technology Readiness Level) of the various components, to the innovative way in which they can be integrated, quantified by the IRL (Integrated Readiness Level) and SRL (System Readiness Level), and to the new way of operating these systems.

In this complex scenarios, new players are emerging, paving the way for the commercialization of spaceflight, aiming at increasing the public consensus in this field of research, keeping the attention high on these projects and arising funds for whatever kind of initiatives with the ultimate aim of promoting hypersonic flight (Peeters, 2003). Among them, different ventures are currently on-going or programmed for the next few years, devoted to offer touristic services to live an astronaut-like experience (Crouch, 2001), feeling the emotion of floating in space and looking at the Earth curvature from an altitude of about 100 kilometres. This kind of projects can be considered the initial step of an incremental path (Figure 2) leading the scientific community to push forward the limits of technologies and developing a pretty new, fast, man-rated, reliable, reusable and safe transportation system able to go through the atmospheric layers and beyond, redefining the borders of the overall aerial domain.

Since some years, microgravity experiences are offered by aeronautical companies, carried out using commercial aviation vehicles (Gilles, 2008) and one-life experiences on board military vehicles (Studer, 2011) (for both touristic and scientific purposes) are sold too. Starting from the lessons learned in these contexts, ad-hoc transportation vehicles will be developed, built and test, at first in unmanned version, then manned versions, with a long-term desire of providing not only a one-life experience but a routine transportation system to be a competitor on the civil aviation market.

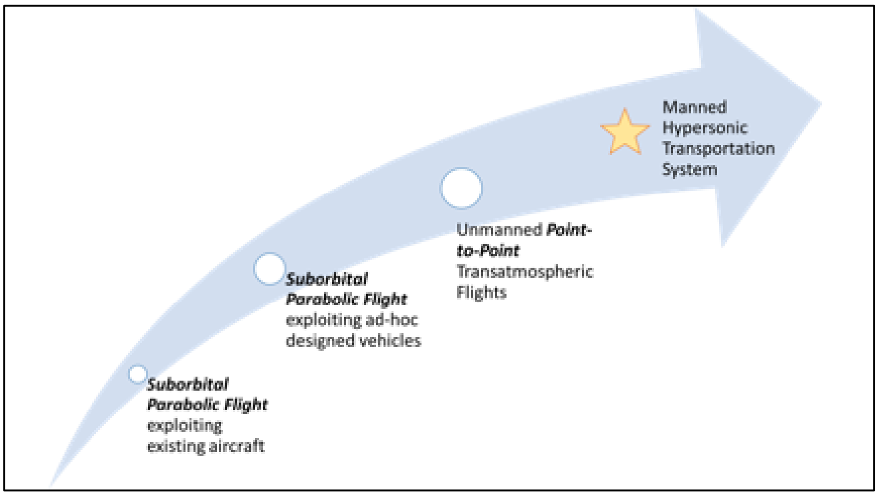


Figure 2: Incremental path for the development of a man-rated Hypersonic Transportation System

In this highly competitive scenario, the role of the leader engineer in charge of the conceptual design of the new transportation system is crucial. In the past

(McMasters, 2002), this role was committed to the most aged and expert engineer of the company or of the research centre who was considered like a druid of the ancient community. Taking experience from all the previous lessons learned and using inspiration (sometimes from past projects in the same or different transportation fields, in other cases directly from nature) the designer was able to start sketching and proposing new products. This way of thinking and designing allows to create completely new shapes and architectures, permitting to have a very high variety of products, each one with its own peculiarities. However, this process, that was acceptable at the beginning of the civil aviation era, cannot be anymore applicable for different reasons. First of all, the initial economic effort is currently too high to allow engineers playing with disrupting innovative shapes without having any demonstrated superiority in performances with respect to the other competitors on the market. Besides any economic reasons, that approach is no longer applicable considering the current restrictions in the development time, implying very fast iterations during the design process. Moreover, time and budget constraints force to postpone, as much as possible, the decision point for the baseline selection until having enough elements to diminish the risk of selecting a non-optimal alternative. Of course, there are many other reasons for which a change in aerospace design activity is required and they will be clear in the next chapters, but these are sufficient to justify the need of developing an innovative integrated methodology. This approach should deal with very complex aerospace systems and with the primary aim of supporting the designers in developing new **highly competitive** products (Raymer, 2006). To discover the key elements of the new design methodology, it is convenient to answer the question: “How can a new aerospace product be competitive in the current market?”. Probably, the most exhaustive answer is that a product should reflect all the stakeholders needs and desires in the best possible way, being careful and wise in making the best trade-off in case of really antithetical requirements. Thus, it is necessary to clearly define the class of stakeholders directly or indirectly involved in the project and then in depth analyse their role in the society and their way of thinking (especially if they come from abroad or if one is dealing with international organizations). All the categories of stakeholders directly related to the product, like funders, airlines, crew, maintainers should be considered in these analyses. Moreover, it is really important to widen this group including also all those people that will directly exploit the service offered by the product (passengers, scientists, etc…) or all those inhabitants that could receive benefits indirectly or that are exposed to a higher risk of incidents. For this reasons, “be competitive on the market” does not only mean to be the best one in terms of performances but it means be the best in terms of performances, guaranteeing an adequate level of **environmental compatibility** and **safety**. Guaranteeing environmental compatibility is not only related to the pollutant emission but it is a broader concept that considers all the interfaces and related problems of an artefact inserted in an inhabited natural environment. Thus, it takes also into account additional issues like the disposal of the different materials, the pollutant emission footprint and the acoustic emissions (e.g. sonic boom problem). As far as safety is concerned, it is important to notice that it is not only referred to guaranteeing the survivability of the passengers but it consists also in all those practices devoted to diminishing the risk of incidents on ground (directly or indirectly due to the aerospace product) or in the air, with respect to the other airspace users. This means that it is no more sufficient to think about the aerospace product without conceiving, since the beginning, the way and the environment in which it will be operated and all the supporting infrastructures that will make possible its mission. For this reason, in this Thesis, the author will refer to the transportation system as fully integrated within a complex System of System (SoS). From the technical point of view, a proper methodology supporting a fully integrated traceability should be envisaged, in order to check the fulfilment of the initial stakeholders' needs all along the project. For this reason, a Model Based System Engineering (MBSE) approach has been selected to be used.

Summarizing, the present work aims at proposing a methodology for the conceptual design of a highly competitive, environmental friendly and safe transportation system, suggesting a tool-chain to support the process enhancing the traceability and the cost-effectiveness of the design process. In addition, for the sake of clarity, hypersonic transportation systems are used as examples for the application of the methodology. Please notice that in order to maximize the homogeneity of the text and to ease the comprehension, each chapter contains both the description of the suggested theoretical approach and a related example of application or considerations focused on the design of a hypersonic transportation system.

Chapter 1 aims at providing the readers with an overview of aerospace initiatives carried out in the field of hypersonic, of the typical reference missions and of the main configurations enviable for hypersonic transportation systems. Furthermore, information about existing and under-development spaceports are reported.

Chapter 2 provides a step-by-step description of the conceptual design methodology presented in the Thesis and outlines the general architecture of a support tool-chain.

Chapter 3 focuses on Mission Analysis (MA) and Concept of Operations (ConOps) definition. The Mission Analysis starts from the identification of all the stakeholders, their categorization and the detection and proper formulation of their needs and desires. In parallel, it is also important to consider market forecasts and identify additional non-aerospace domains where some innovative technologies could be exploited, guaranteeing additional economic benefits. Considering the high level of innovation related to this kind of products, it is important to take into account regulations and in cases (like hypersonic transportation) in which they are not available yet, it is important to define a cooperation plan between designers and legislators. At this point the methodology will lead to write down the mission statement and the derivation of the first high level mission objectives and the elicitation of the first draft list of mission requirements and constraint. Starting from this list, the methodology leads to the identification of the highest possible number of mission architecture alternatives, providing a rationalization of the selection process reducing the subjectivity of the trade-off analyses. The overall process described in Chapter 3 can be fully implemented in a MBSE approach guaranteeing the traceability of stakeholders' requirements within mission objectives, in the first list of requirements and also in the identification of the best mission architecture. Complementary, the second part of the Chapter is mainly related to the identification of preliminary feasible trajectories for the transportation system, highlighting mission phases and timelines in both nominal and out-of-nominal conditions. In this context, options to enhance safety at SoS level for a hypersonic mission are suggested. Moreover, it is also important to provide a sketch of possible communication strategies, layout of possible on-ground sites able to host and support the system, to verify if the service will be executed by a single vehicle or a fleet will be envisaged, impacting on the sizing of both logistics and maintenance infrastructures and personnel. At the end of the chapter, a brief section about project management and roadmapping activities suggesting support to the definition of medium and long-term strategic plans.

Chapter 4 enters in the detail of the design of an innovative and competitive transportation system. In particular, this chapter provides an overview of the design and sizing process, highlighting the major complexities from different standpoints. The analysis starts from the identification of the major challenges the designer face in identifying the optimal layout, considering the aerothermodynamics and structural integration. Then, a proper section highlights the major challenges related to integration of innovative and multifunctional subsystems (e.g. Thermal and Energy Management System). Eventually, the influence of trajectory definition on the design of hypersonic vehicles is focused.

Chapter 5 is the first strictly related to the design and size of the transportation system. In particular, it deals with wing design and sizing providing elements for selecting the best layout for the identified application. In addition, an investigation of advantages related to the mutual position of wing with respect to the fuselage for the specific case of hypersonic is carried out.

Chapter 6 deals with the conceptual design of the fuselage, with special attention devoted to the design of high-tech passengers' compartments and to possible strategies to increase safety of crew and passengers with a proper design of the fuselage. In this context, the conceptual design of an escape system for a vehicle aimed at performing suborbital flight is presented.

Chapter 7 deals with the problem of integration. This Chapter starts with suggestions to define wing and fuselage structure during conceptual design phase and to integrate these two major elements in an optimal way. Then, additional considerations related to the integration of propulsive system, landing gear and empennages with related control surfaces definition are addressed. Eventually, an example of multidisciplinary tool-chain relating software for requirements management with CAD and dynamic simulation software is shown.

Then, conclusions are drawn suggesting possible ways of widening the methodology and the tool-chain considering more carefully additional aspects such as costs and aerothermodynamics analyses. Moreover, additional suggestions for the development of an feasible and optimized hypersonic system are suggested.

References

Crouch, G. I. (2001). The market for space tourism: early indications. Journal of Travel Research 40.2 (2001): pp. 213-219.

Gilles C., et al. (2008). Perception of depth in microgravity during parabolic flight. *Acta Astronautica* 63.7 (2008): 828-832.

Jakhu, R. S., et al. (2011). The Need for an Integrated Regulatory Regime for Aviation and Space: ICAO for Space?. Vol. 7. *Springer Science & Business Media*.

McMasters, J. H., et al. (2002) Airplane design-past, present, and future." *Journal of Aircraft* 39.1 (2002): 10-17.

Peeters, W. (2003). Space commercialization trends and consequences for the workforce. *Acta Astronautica* 53.4 (2003): 833-840.

Raymer, D. P., (2006). Aircraft Design: A Conceptual Approach and Rds-Student, Software for Aircraft Design, Sizing, and Performance Set. *AIAA Education. AIAA (American Institute of Aeronautics & Ast, 2006.*

Studer M., et al. (2011). Parabolic maneuvers of the Swiss Air Force fighter jet F-5E as a research platform for cell culture experiments in microgravity." *Acta Astronautica* 68.11 (2011): 1729-1741.

Chapter 1

Dealing with hypersonic

This chapter aims at providing the readers with an overview of the major initiatives in the development of hypersonic transportation systems from both an historical and a technological standpoint. Considering the currently under development initiatives and the results of the latest market analyses, potential mission scenarios and possible vehicle configurations are suggested. At the end of the chapter, special attention is reserved to the description of the other elements of the System of Systems in which the vehicle is supposed to be operated, such as spaceports that will play a fundamental role.

1.1 An historical perspective

Accepting the definition of Kenneth Chang, October the 20th, 2014 on the eminent New York Times (Chang, 2014), ***spaceplanes*** *are so fascinating because they are aerospace vehicles able to operate as aircraft when they are in the lower atmospheric layers and as spacecraft when they are in space*. From one side, this definition is essential to understand the reasons of such increasing interest in this vehicles by both the aeronautical and space domains, but on the technological point of view, it reveals the level of complexity of such a transportation system.

It is clear that spaceplanes should be capable of going hypersonic. Think about all the re-entry missions that involves capsule instead of high lift-over-drag ratio (L/D) vehicles. But configurations closer to aircraft are the only one that could guarantee reusability, higher manoeuvrability and comfort for passengers.

Moreover, being able to fly in the aeronautical airspace, they should be more easily integrated as far as procedures and regulations concern.

The current positive interest in developing spaceplanes reached its main push in 2004, when the first private spaceflight became reality, officially paving the way to the touristic suborbital flights.

On October 4th, 2004, at Mojave Airport, a civilian test centre in the middle of the desert in California, about 150 km from Los Angeles, ***SpaceShipOne*** (SS1) and its carrier, named WhiteKnight (Figure 3), completely conceived and developed by the private company Scaled Composite founded by Burt Rutan in 1982. This project was one of the participants to the competition announced by X-Prize Foundation some years before. And one of the major exited and inspiring speech just after the succeeding mission of SS1 was given by the co-chairman of the foundation (Seedhouse, 2014):

"*Today we have made history. Today we go to the stars. You have raised a tide that will bring billions of dollars into the industry and fund others teams to compete. We will begin a new era of spaceflight*." (Peter Diamandis)

Figure 3: SpaceShipOne on top of its carrier aircraft, the WhiteKnight

It is really curious to notice that only a few kilometres away, at Edward Air Force Base, in 1963, the test pilot Joe Walker reaches for the first time the edge of space by flying an Air Force ***X-15*** rocket plane (Figure 4), overcoming the limit of 100 km that is considered the limit between air and space domain as suggested by the International Aeronautical Federation (FAI), the worlds governing body for aeronautics and astronautics.

X-15 was the most famous vehicle coming out from the American X-Series program, undertaken by the National Advisory Committee for Aeronautics (NACA). X-15 design started in 1954 aimed at becoming the first manned aircraft to perform a hypersonic flight. After being dropped by the carrier B-52 aircraft, thanks to the thrust guaranteed by the burn of the rocket engine, it was able to reach the target altitude of 100 km and then it performed an unpowered landing in the same base from which it took off (Guthrie, 1988), (Hannigan, 1994). The X-15 is remembered for many firsts: it was the first aircraft to fly hypersonic (Mach 6.7 during its maiden flight), it was the first manned vehicle to ascend above the limit of atmosphere and space (reaching 107 km of altitude) and it was an incredible testbed for very innovative technologies, especially the propulsive ones (notice that some of the tests aimed at utilizing the more recently developed supersonic combustion ramjet or scramjet air-breathing propulsion systems) allowing the development of the Hypersonic Research Engine (HRE), managed by the NASA Langley Centre. Three X-15 vehicles have been developed and flew 199 times among 1959 and 1968 (Hannigan, 1994), demonstrating that spaceflight could be performed on "regular" basis. In 1994, a cost estimation of the X-15 program was estimated in about US$ 1 billion.

Figure 4: X-15

Figure 5: X-20 Dyna-Soar

Like in the case of SS1, the success of the X-15 gave an incredible push in the development of new vehicles to overcome the limits. Indeed, it was followed by the advent of the ***X-20 Dyna-Soar*** (Figure 5) aimed at demonstrating that the same mission could be flown developing a winged and piloted vehicle launched on the top of a modified version of the Titan 3, a famous InterContinental Ballistic Missile (ICBM). The name of the vehicle itself refers to the characteristic of dynamic soaring envisaged for the configuration. Indeed, differently from the parabolic profile executed by the X-15, the X-20 seemed to be aimed at demonstrating the concept of atmospheric skipping and hypersonic glide at very high altitudes. Unfortunately, the programme was suspended and cancelled by the American government that at that time had to fund in parallel two manned programs. This was probably the main reason for the stop although, at the end of the '90s, engineers confirmed that the major challenge they faced with was the fact that X-20 required a new material that they evocatively defined "unobtainium". The program costed approximately US$ 1.8 billions (as evaluated in 1994) (Hannigan, 1994).

Among the 1960s and 1970s, before the development of the Space Shuttle (officially started in 1972), all over the world, the attention was devoted to the development of enabling technologies. In particular, several ***Boost Glide Re-entry Vehicles*** (BGRVs) allows to cover unexpected distances of more than 25.000 km of range and among them, the two lifting bodies ***ASSET*** (Figure 6) and ***PRIME*** (Figure 6) played an important role in the definition of the major aerodynamic characteristics of hypersonic flight. Moreover, ***M2-F2***, ***HL-10*** (Figure 7) and ***X-24A***, ***X-24B*** and ***X-24C*** (Figure 8) were developed to test the possibility of re-entering from orbital speed and altitudes. In particular, the X-24C was a clear example of technology – pushed design. Indeed, it was designed to properly integrate scramjet with the vehicle (Mach 8 was the target). Between 1965 and 1969, a wide range of configurations were developed within the Aerospaceplane program that before being cancelled, was clearly stating that at the moment, air-breathing engines were not ready to face with hypersonic flight and US Department of Defence (DoD), decided move to the examination of rocket technologies. In that moment, the initiative of the Boeing Aerospace Corporation developed the ***Reusable Aero Space Vehicle*** (RASV). It was intended to be a piloted, fully reusable, rocket only, SSTO vehicle.

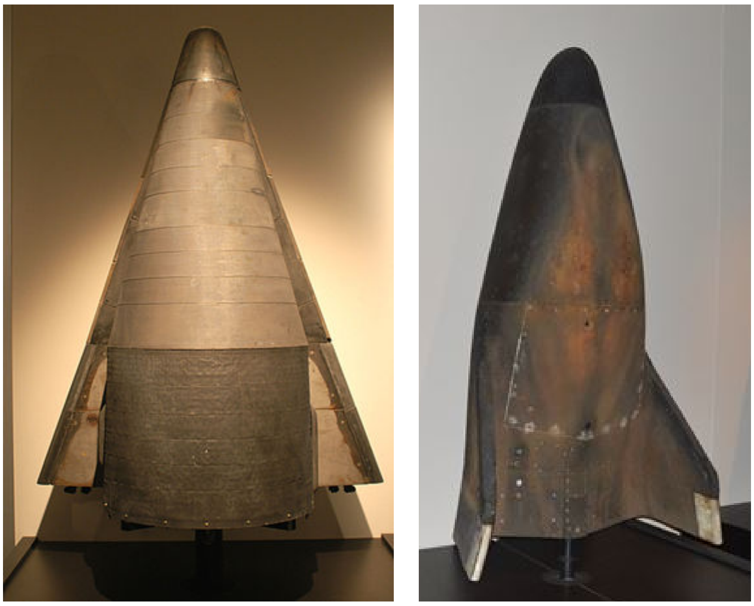


Figure 6: ASSET (left) and PRIME (right)

Figure 7: HL-10

Figure 8: X-24 A (left), X-24 B (centre) and X-24 C (right)

Meanwhile, Considering the European contest, and in particular the United Kingdom, in those years, the British Aircraft Corporation developed **Mustard**, a pure rocket, vertical take-off reusable launcher involving the use of three large lifting bodies stacked together with a special inter-stages propellant management strategy (Wilson, 1986)

On the same years, in Germany, Eugen Sängers, considered the father of the spaceplane with its Silbevogel, developed the well-known ***Sänger I*** (Figure 9), a Two Stage To Orbit configuration. Later in the 1960s, Germany promoted a test program called ART aimed at launching reusable BGRVs on top of a European rocket from an Australian base. Also in this case, the program was terminated after the test phases in 1973 (Kuczera, 1992).

Also the Soviet Union was involved in the rush to hypersonic and the most significant initiative in the early 1960s was the **50/50 concept** (Figure 10), a curious lifting body launched off the back of its carrier aircraft during a hypersonic flight phase (approximately Mach 6).

Figure 9: Sänger I

Figure 10: Soviet 50/50

Figure 11: Space Shuttle

The 1970s in America were literally signed by the ***Space Shuttle*** (Figure 11) program. One of the best description of the goals defined at the beginning of the project was told by George P. Miller, the chairman of the former House Committee on Science and Astronautics, on April 23rd, 1970 (Hannigan, 1994), (Hechler, 1982):

The key to success of this Nation's future space efforts lies in the development of a low cost, recoverable, and reusable space transportation system. The reusable Space Shuttle will drastically reduce the cost of putting people and cargo into space. In particular, the Shuttle will facilitate construction of a manned orbiting Space Station that will open up new areas of scientific and technological activity in the near neighbourhood of Earth.

*(*George P. Miller*)*

Unfortunately, during phase B study, two serious problems arose with the fully reusable configuration. The fist was a technological concern over whether this vehicle could actually be built, considering all the difficulties of integration of liquid hydrogen and liquid oxygen tanks into the orbiter. The second reason was strictly related to funding: at the end of phase B, the estimated costs to the completion of the projects were much higher than expected. Thus, the original optimal configuration of the Space Shuttle that at the beginning consisted in two fully reusable stages, was gradually whittled down reducing technological risk and costs. In particular, the Shuttle redesign in 1971 reduced the estimated development costs from more than US$ 10 billion to US$ 5 billion and NASA chose this configuration as baseline with the bless of President Nixon (January 5th, 1972) (Allaway, 1979):

I have decided today that the United States should proceed at once with the development of an entirely new type of space transportation system designed to help transform the space frontier of 1970s into familiar territory, easily accessible for human endeavour in the 1980s and 1990s.

(President Nixon)

Considering that the original goals of the Space Shuttle, mainly in terms of reusability, would not have been completely realized, since the beginning of the 1980s, the American Air Force started the ***Trans-Atmospheric Vehicle*** (Figure 12) Program (TAV) which was later replaced by the NASP program. The concept proposed an air-launched small orbiter supplied by a discardable droptank. In its first concept, it should be air-launched by a Boeing 747 and this was incredibly similar to the Soviet's **MAKS** that was envisaging to use an Antonov AN-225 as carrier.

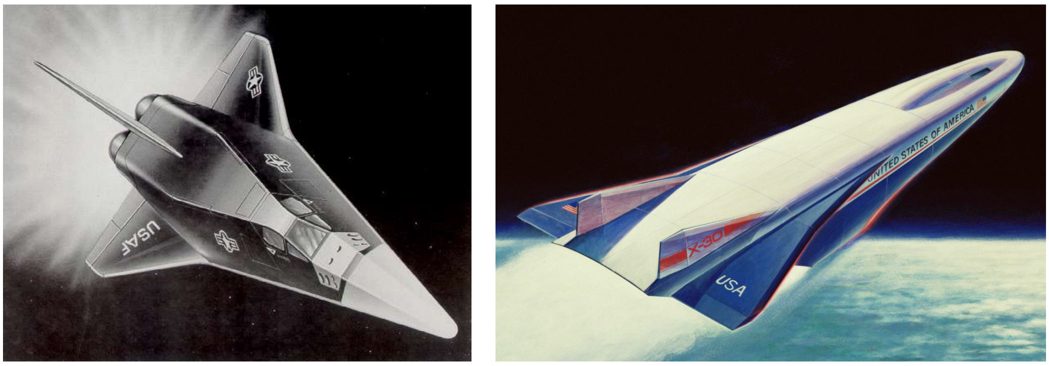


Figure 12: TAV (left) and NASP X-30 (right)

The ***National Aero-Space Plane*** (NASP) (Figure 12) received the official governmental support by President Regan that on February 4^{th}, 1986 made this announcement, paving the way to the design of the first Point-to-Point transportation system:

We are going forward with research on a new Orient Express, that could, by the end of the next decade, take off from Dulles airport and accelerate up to 25 times the speed of sound, attaining low Earth orbit or flying Tokyo within two hours

(President Regan)

This statement clearly fixed the main goal of the NASP program: to demonstrate the feasibility of sustained hypersonic cruise, but the very primary impetus was to develop an experimental vehicle capable of flying directly to low Earth Orbit. Since the beginning, it was crystal clear that the combined requirements of Single Stage To Orbit through breathing air at the greatest possible speed is an immensely challenging goal. Additional innovative element of this projects were summarized by the former NASP Program Director:

The goal of the NASP program is to develop and demonstrate the feasibility of horizontal take-off and landing aircraft that utilize conventional airfields, accelerate to hypersonic speeds, achieve orbit in a single stage, deliver useful payloads to space, return to Earth with propulsive capability and have the operability, flexibility, supportability and economic potential of airplanes. In order to achieve this goal, technology must be developed and demonstrated which *is real a quantum leap from the current approaches being utilized in today's aircraft and spacecraft.*

(Dr. Robert Barthelemy).

Thus, it was in the 1980s that the need of developing transportation systems aimed to be competitive with both aircraft and spacecraft become a push for the research in the spaceplane field. The experimental vehicle that may eventually emerge from NASP effort has been designated as X-30. Although the cancellation at the beginning of the 1990s, the NASP program accelerated the technology to the point where the feasibility of developing spaceplanes might be feasible.

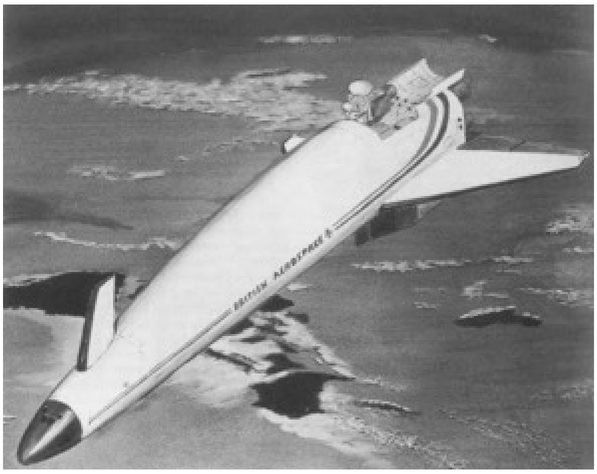


Figure 13: HOTOL

During the same years, the ***HOrizontal Take Off and Landing*** (HOTOL) (Figure 13) project was under development in the United Kingdom aimed at significantly reduce the cost of access to space (Parkinson, 1990)0. In an historical and technical standpoint, it has to be notice that the HOTOL arguments have helped focus the attention on the cost-effectiveness of European space programs, bringing attentions to some issues which some European Nations preferred to avoid. In some ways, it is possible to affirm that the emergence of HOTOL was indirectly responsible for the German Sänger program and the ESA Future European Space Transportation Investigation Program (FESTIP). Initially developed as the result of the independent thinking of Dr. Robert Parkinson and Alan Bond, the program aimed at designing a vehicle as small as possible and able to host from 7 to 10 tons of payload. Thus, HOTOL was conceived as an unmanned vehicle because the mass of a crew compartment sufficiently large to host 6 passengers, could be easily closed to the payload itself. Moreover, as it is clearly described in the next chapters, the integration of a crew compartment into the vehicle design shall be taken into account since the beginning of the design because in many cases, servicing activities can become more complex to ensure that the crew compartment and the vehicle could maintain the same safety levels in the following missions. HOTOL was never intended as a technology or experimental program for its own sake, but was to lead directly to an operational launch system. Unfortunately, the program was not properly supported by politicians and this led to its downfall but the vision of a HOTOL-like capabilities remained.

In the same years, within Europe, we assist to some examples of detrimental effects of no-cooperation among Nations. In particular, within ESA context, until the mid of the 1980s, France was the undisturbed playmaker of the space agency while Germany's role was more prominent within the Ariane consortium. Mainly for this reason, Germany wanted to provide about the same funds to Hermes as France intended to provide and also wanted to integrate one of the two Hermes vehicles in its own facilities. Essentially, Germany wanted to benefit from Hermes Technology. Considering its role, France refused to allow the participation in the technology aspects that are of specific interest to Germany, like the re-entry system. With this background, reducing cost to orbit, the development of HOTOL, responding to French dominance and the need to push aerospace technology, the Hypersonic Technology Program was approved by the German government and the ***Sänger II*** (Figure 14) concept was conceived. Originally, it was articulated in three main phases. The first phase aimed at performing extensive conceptual design studies, with a special effort devoted to the feasibility evaluations of the air-breathing first stage. This is also the phase during which the main test infrastructures should be built. The second phase would be opened to international collaborations aimed at performing the preliminary and detailed design. The result of the phase should be the construction of a demonstrator. Depending on the results of this intermediate but crucial phase, the third one should be devoted to the prototyping of the two stages. The basic idea was not to provide a high-speed passenger airline service but to create a space transportation system following a peculiar mission profile, that deserve to be described in detail (Koelle, 1990). Sänger takes off exploiting its own landing gear from a runway (supposed to be in located in a remote region of Central Europe) and accelerate using turbojet engines up to 10 km of altitude. At this moment, the ignition of the afterburners allows the vehicle to reach the supersonic speed. After Mach 3.5 is reached, the propulsion system moves from turbojet to ramjet. The highly integrated propulsion system allows the vehicle to perform a cruise at more than 25000 km of altitude at Mach 4.4. The range to be covered strictly depends on the orbital parameters in the moment of the launch. While approaching to the desired latitude, Sänger turns due east and the ramjets are used to reach an altitude of 31 km and Mach 6.8. Please note that this would be the moment in which the highest temperature would be experienced. In this condition, a pull-up manoeuvre of 6 degrees is executed allowing to increase the altitude up to 35 km. At this point, the stage separation takes place just after the second stage is cranked up on its three main point mounting to an angle of about 2 degrees with respect to the first stage and the two small orbital manoeuvring engines are ignited. Some seconds after the release, the second stage main rocket engine is throttled up allowing the vehicle to ascent to its initial transfer orbit that is later on circularized using again the orbital manoeuvring engines. In parallel, the manned first stage flies back to the launch site for servicing. For the second stage, both a cargo and a manned version have been envisaged, simply replacing the cargo bay with a crew compartment. On the completion of the mission, which lasts no more than 50 hours, the second stage fires its two orbital manoeuvring engines to change orbit for re-entry. Moreover, in order to allow the landing in the same Central Europe runway, the vehicle shall re-enter in a corridor permitting it to fly 2700 km north of its orbital ground track. This high cross range trajectory issue was overcome hypothesizing a skip –glide trajectory that could also benefit of sequential re-radiating heat that allows to limit the thermal loads if compared with a direct orbital re-entry. At the end, the second stage perform a glide approach an unpowered but controlled landing. The need of developing two different vehicles (the two stages separately) deeply impacted on costs that in 1987 were estimated in about US$ 14.5 billion (GAO, 1991). Sänger remains a lead concept acting as a focus for the German Hypersonic Technology Program.

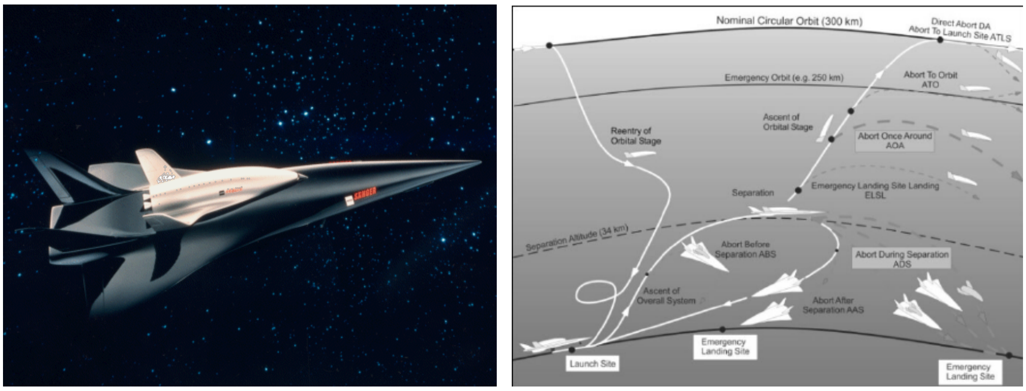


Figure 14: Sänger II and its mission profile

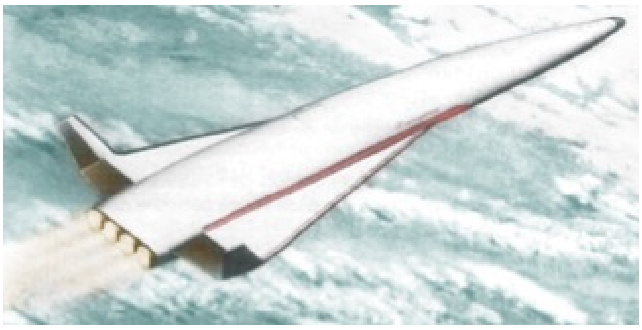


Figure 15: STS-2000

Figure 16: MAKS

On the other side, France has been interested in high-speed air-breathing flight as long as any of the other nations and in collaboration with UK, in 1969 they succeeded in building the Concorde, the Mach 2 airliner. Looking at hypersonic, the most interesting initiative were the ***Aerospatiale STS-2000*** (Figure 15) and the ***Star-H*** concepts. These first studies led France to the PREPHA program, focused on 5 main research areas: propulsion system, CFD numerical techniques, materials, vehicle subsystems and modifications of the already existing test facilities.

Considering Soviet Union, the most brilliant initiative related to hypersonic transportation was the development of the ***MAKS*** (Figure 16) vehicle concept, based on the availability of Antonov AN-255 as carrier aircraft. The strange configuration was achieved thinking at the main goal of enhancing the space station logistic support and both a manned and an unmanned mission have been proposed.

Also in Japan, many activities related to hypersonic transportation systems have been carried out. In particular, the Japan National Aerospace Laboratory put lot of efforts in the definition of a ***SSTO*** designed as test bed for the LACE propulsion technology. It was aimed at placing a crew of two members and 8 passengers into a space station in LEO. The noticeable dimensions of the vehicle pushed some Japanese officials to think about a different system, a purely experimental version of HOPE.

Coming to the last two decades, an higher interested Stakeholders desired to finalize some projects. In this context, an even higher number of different conceptual design activities as well as technological advancements have been carried out all around the world and many of them are related to the commercialization of spaceflight and to the increase of the touristic flights demand. In particular, as it is detailed in the following section, suborbital missions allowing to offer a unique experience to tourist but also to be the place for testing technologies, have been elected to be the most feasible mission exploiting the current technologies and with a more limited available budget. Among the various initiatives, it could be interesting to consider the several concepts proposes to compete the Ansari X-Prize that has been won by Scaled Composite. The X PRIZE Foundation was established in 1994 as an educational, non- profit corporation dedicated to inspiring private, entrepreneurial advancements in space travel (AACST, 2001). The St. Louis-based X PRIZE Foundation was offering a $10 million prize to the first team able to launch a vehicle capable of carrying three people to a 100- km sub-orbital altitude and repeating the flight within two weeks (only one person and ballast for two others are required to actually make the flights). The X PRIZE was offering to help speed along development of space vehicle concepts that will reduce the cost of access to space and to allow human spaceflight to become routine.

The X PRIZE (Table 1) competition had 17 entrants from five countries offering a variety of different RLV concepts. The commercial vehicles under development for the X PRIZE competition are uniquely designed for sub-orbital space tourism operations carrying about three to six passengers. These designs use many different take-off, landing, and design concepts, but all plan to use existing technology to accomplish their goals.

Table 1: X-Prize

Program	Developer	Vehicle description
Ascender	David Ashford, Bristol Spaceplanes Limited	Spaceplane powered by two conventional jet engines and a liquid-fuelled rocket engine. The vehicle will take off and land horizontally.
Astroliner	Kelly Space Technology	See next paragraph.
Cosmos Mariner	Dynamica Research	Spaceplane powered by two air-breathing engines and one rocket engine. The vehicle will launch and land horizontally.
Gauchito	Pablo De Leon and Associates	Two-stage vehicle that will launch vertically. The first stage booster and the second stage passenger capsule return to Earth using parachutes.
Green Arrow	Graham Dorrington	Cylinder-shaped rocket using liquid-fuelled rocket engines. The vehicle will launch vertically and land vertically using parachutes and air bags.
Lucky Seven	Mickey Badgero	Cone-shaped vehicle powered by rocket engines. The vehicle will launch vertically

		and land using a parafoil.
Mayflower (CAC-1)	Advent Launch Services	Cylinder-shaped glider powered by liquid-fuelled rocket engines. The vehicle will launch vertically from water and land horizontally in water.
MICHELLE-B	TGV Rockets	The vehicle will launch vertically and land vertically using ascent engines in a deep throttle mode.
PA-X2	Rick Fleeter, AeroAstro Inc	Cylinder-shaped vehicle using a liquid-fuelled engine. The vehicle will launch vertically and land horizontally using a steerable parafoil.
Pathfinder	Pioneer Rocketplane	See next paragraph.
Proteus	Burt Rutan, Scaled Composites	Two-stage vehicle consisting of the conventional turbo- fan powered Proteus aircraft and a rocket-powered second stage.
The Space Tourist	John Bloomer, Discraft Corporation	Disc-shaped vehicle powered by air-breathing “blastwave-pulsejets.” The vehicle will take off and land horizontally.
Thunderbird	Steven M. Bennett, Starchaser Foundation	Cylinder-shaped rocket using air-breathing engines and liquid fuelled rocket engines. The vehicle will launch and land vertically.
X Van	Pan Aero, Inc., Third Millenium Aerospace	Pan Aero has publicized two designs for the X Van. The entry may be a two-stage-to orbit system comprised of a booster stage and orbiter stage, or a single-stage system flying a sub-orbital trajectory.
unnamed	William Good, Earth Space Transport System Corporation	Not public available
unnamed	Cosmopolis XXI	Cylinder-shaped rocket which is launched off of carrier aircraft “Geophisika”. The vehicle will take off vertically and land horizontally.

Also the American continued to develop both commercial and governmental programs (AACST, 2001). Among the most famous American commercial programs, ***Astroliner, Rocketplane, Space Access-1*** (Figure 17)***, Space Cruiser System*** and the famous ***VentureStar*** (Figure 18) could be remembered.

Astroliner has been designed on a proprietary patented tow-launch technique. In particular, it should be towed into the air by a modified Boeing 747 aircraft to an altitude of 6 km where the 38-meter long RLV will be released and proceed on a suborbital trajectory under its own power. Astroliner will then use expendable upper stages to inject payloads into orbit. Following separation from the tow aircraft, the Astroliner ignites its rocket engine(s) and climbs to an altitude of 125 km and a speed of Mach 6.5. The nose of the vehicle then opens to release the upper stages and payload. The Astroliner then re-enters the atmosphere and returns to land at a conventional airfield under the guidance of its two-pilot crew. The Astroliner design also features wing-mounted jet engines for powered descent and landing.

Concerning the RocketPlane, the so called “Black Horse” spaceplane, was promoted within the United States Air Force in the early 1990s. Pioneer Rocketplane renamed the vehicle “Pathfinder” and proposed it as a potential design for NASA’s X-34 program. Although the Pathfinder was not selected for the X-34, the company elected to continue Pathfinder development. Pathfinder is a spaceplane operated by a crew of two and is powered by both air-breathing jet engines and LOX/kerosene rocket engines. The vehicle takes off horizontally using turbofan jet engines following a very aeronautical procedure. When it reaches an altitude of 6 km, it docks with a tanker aircraft allowing Pathfinder’s LOX tanks in a method identical to air-to-air refuelling. After disconnecting from the tanker, the spaceplane ignites its rocket engine and climbs to an altitude of 112 km at a speed of Mach 15. Once out of the atmosphere, it can open its cargo bay doors and release its payload with a conventional liquid rocket upper stage. The payload is then carried into orbit as the spaceplane re-enters the atmosphere. After deceleration to subsonic speeds, the vehicle should be able to switch on its jet engines and perform a powered horizontal landing.

The Space Assess SA-1 has been proposed to conduct satellite launches by Space Access LLC. The concept consists of an unpiloted spaceplane that uses a hybrid propulsion system and one or two rocket- powered upper stages to deliver a full range of payloads to LEO or Geostationary Transfer Orbit (GTO).

The Space Cruiser System (SCS) vehicle is being designed and developed by Virginia-based Vela Technology Development, Inc. to carry six passengers on a sub-orbital flight reaching an altitude of just over 100 km. SCS is a two-stage horizontal-take-off and landing design that employs both air-breathing and rocket engines. The first stage booster, or “Sky Lifter,” will be piloted by a two-member crew and will be powered by two jet engines. The Space Cruiser second stage spaceplane will be carried underneath the Sky Lifter. he two stages will climb together to about 15 km where the Space Cruiser, carrying two crewmembers and six passengers, separates and will climb to 100 km using its three rocket engines. During re- entry into the atmosphere, the Space Cruiser will fire retro-rockets to slow the vehicle’s descent and then will activate two turbo-jet engines to return to a landing site performing a controlled and powered re-entry.

VentureStar is Lockheed Martin's potential commercial follow-on to the X-33 vehicle being developed for NASA's RLV program. Its main characteristics is that it will be powered by seven linear aerospike engines.



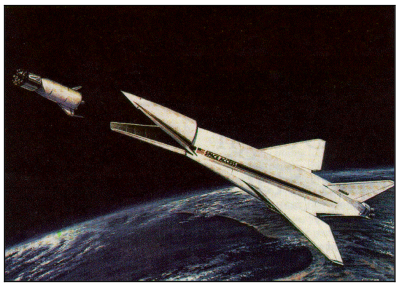


Figure 17: RocketPlane (left) and Astroliner (centre) and SA-1 (right)


Figure 18: Space Cruiser System (left) and Venture Star (Right)

Moving to the American Government Programs, the continuation of X-Series should be mentioned. In particular, ***X-33, X-34, X-38 Crew Return Vehicle*** (Figure 19)***, X-40 A*** and the ***X-37*** (Figure 20).

Starting with the X-33, it was targeted to reach high hypersonic speeds and demonstrate SSTO and autonomous operations capabilities and to reach this goal a special focus have been reserved to the test of new technologies, especially the propulsive one.

The X-34 project started in 1994, again with the aim of demonstrating key technologies with the goals to achieve a maximum speed of Mach 8 and to reach altitudes of up to 80 km Following a competition that included nine entries, NASA awarded Orbital Sciences a $60 million contract in June 1996 to design, develop, and test the X-34 that was then moved in the development of Pathfinder in 1999.

A completely different idea led the design of the X-38. It is a technology demonstration vehicle project of the Johnson Space Center and Dryden Flight Research Center and can be considered a prototype for a crew return vehicle (CRV) that will be attached to the International Space Station.

Differently to the previous ones, the X-40 it is a clear example of military spaceplane demonstrator. In particular, it is a multimission vehicle able to perform a variety of orbital and suborbital military missions. In particular, the concept consists of a reusable “mini-spaceplane,” or Space Maneuver Vehicle (SMV) that is carried to hypersonic speeds by a sub-orbital reusable first stage. The SMV is released and accelerates to orbit, where it will be designed to manoeuvre in space and remain on-orbit for perhaps as long as one year.

On July 14, 1999, NASA and Boeing signed a cooperative agreement to develop a Future X Pathfinder vehicle designated X-37. The X-37 is the first NASA X vehicle that is designed to operate in both orbital and re-entry phases of flight, and it will be the fastest of the vehicles designed to reach speeds of up to Mach 25. Once deployed, the X-37 will remain on-orbit for up to 2 days on the first mission and up to 21 days on the second mission. Once on-orbit, the X-37 will test space vehicle technologies, including a solar array system developed by the Air Force. The vehicle will re-enter the Earth's atmosphere and land autonomously on a conventional runway (Boeing, 1999).

Figure 19: X-34 (left) and X-38 (right)

Figure 20: X-40 (left) and X-37 (right)

Also in these last decades (Tauri Group, 2012), other important initiatives have been carried out all around the world. For example, at the end of the 1990s, India's Defense Research Development Organization (DRDO) announced the initial funding of the design of a small reusable spaceplane. The ***Aerobic Vehicle for Advanced Trans-Atmospheric Research*** (AVATAR). The AVATAR would take off horizontally using ramjet engines that burn air and hydrogen. Once at a cruising altitude, the vehicle would use scramjet propulsion to accelerate to Mach 7. During these cruising phases, an on-board system will collect air from which liquid oxygen will be separated. The liquid oxygen collected then would be used in the final flight phase, when the rocket engine burns the collected liquid oxygen and the carried hydrogen to attain orbit (Hyndustan Times, 1998).

In Europe, different initiatives are currently under development. Among them, the successful mission performed by ***Intermediate eXperimental Vehicle*** (IXV) (Figure 21), February 11th, 2015 has been a real demonstration of the European capabilities of designing, developing and also operating such missions, from the pre-feasibility study (Tumino, 2009), to the system integration, to the vehicle tracking (Tumino, 2008), and the recovery after the splash down in the Pacific Ocean and the under-development post-flight data analyses (Bonetti, 2015)., . In particular, IXV is a re-entry demonstrator whose objective is to tackle the basic European needs for re-entry from Low Earth Orbit (LEO), consolidating the knowledge and expertise necessary for the development of future European re-entry systems. Moreover, it is a technology platform that represents a step forward with respect to the Atmospheric Re-entry Demonstrator (ARD), flown in October 1998, with an increase in-flight manoeuvrability allowing the verification of technologies over a wider re-entry corridor and with the presence of aerodynamic surfaces employed actively to trim and manoeuvre the vehicle during some peculiar phases of the mission profile (Figure 27). ESA is now working on IXV successor, the ***PRIDE*** project.

The Reaction Engine Limited is currently continuing the design and development of the ***SKYLON*** (Figure 21). It is an unpiloted, reusable spaceplane intended to provide reliable, responsive and cost effective access to space. Currently in early development phase, the vehicle will be capable of transporting 15 tonnes of cargo into space. It is the use of Synergistic Air-Breathing Rocket Engine (SABRE)'s combined air-breathing and rocket cycles that enables a vehicle that can take off from a runway, fly directly to earth orbit and return for a runway landing, just like an aircraft.

LAPCAT II (Figure 22) (Long-Term Advanced Propulsion Concepts and Technologies) (Steelant, 2015) is a follow-up of the previous European Community (EC) co-funded project LAPCAT (Steelant, 2008), (Steelant, 2009)The primary objective of the first project was to develop different vehicle concepts enabling the potential reduction of antipodal flight times to about 4 hours along with the identified enabling critical technologies. Among the several vehicle configurations resulted at the end of the research activities, only two concepts characterized by allowing a ***Mach 5 and a Mach 8 cruise flight, were retained for further evaluations in*** LAPCAT II.

Spaceliner (Figure 22) is a project currently under development within DLR. It is a two stage Reusable Launch Vehicle (RLV) aimed at performing Ultra long-haul distances like Europe – Australia in 90 minutes. Other interesting intercontinental destinations between e.g. East-Asia and Europe or the Trans-Pacific-route to North-West America could be reduced to flight times of slightly more than one hour.

Now, having in mind the historical development of the hypersonic transportation systems, it is convenient to go deeper and understanding if it is possible to derive criteria to classify spaceplanes and related missions.

Figure 21: IXV (left) and SKYLON (right)

Figure 22: LAPCAT II (left) and SpaceLiner (right)

1.2 Mission Profiles

The aim of this section is to collect, present and discuss the main mission profiles in which hypersonic speeds could be envisaged or in which the transportation system is supposed to fly at very high altitudes, not commonly exploited in the aeronautical domain. Through the examination of these mission profiles, the main problems to face with are highlighted, pointing out some technical, regulatory or safety lacks.

1.2.1 Parabolic Suborbital Mission Profile

Even if parabolic suborbital mission profiles are not characterized by hypersonic speeds, they are here considered because, (as it has been mentioned in the introduction), they represent the state-of-the-art in the field of innovative transportation system. Parabolic suborbital flight profiles are the most common in the area of touristic and scientific flights aimed at recreating microgravity conditions. In this case the speed regime is just in between supersonic and hypersonic but it is very important to take them into account for several reasons. First of all, suborbital missions are currently under investigations and development by several public and private companies and the market outlooks envisage an increasing demand in the next few years for both touristic and scientific purposes (Seedhouse, 2014). Moreover, this type of mission contains all the features and peculiarities of the other mission profiles presented in this chapter even if, in some cases, with a lower level of required performances. This fact has not to be considered as extremely negative because this is the reason why these missions are considered as the first step in the incremental path towards a hypersonic transportation system. Vehicles and Missions able to perform parabolic flight can be considered as test-bed for enabling technologies in the operative environment, even if with less demanding conditions.

The illustration in Figure 23 sketches a generic mission profile and shows that there are two different possibilities when designing a parabolic mission. Depending on the purposes of the mission and on the available on-ground infrastructures, it is possible to perform a parabolic flight aiming at maximizing the cross-range in a way to connect two different locations on the planet. Playing with parameters such as the angle of the thrust vector, its module and the specific impulse of the propulsive system, it is possible to create a net of connecting parabolic flights. Ballistic missiles can be considered the ancestors of this kind of mission. The famous first ballistic flight, the well-known V-2, was able to cover a maximum operative range estimated in 320 km, reaching a maximum altitude of 88 km and a maximum speed of 5.760 km/h.

In case of missions aimed at touristic flight or manned flight with scientific purposes, the primary interest is no more connecting locations but experiencing a period of microgravity, free-floating for some minutes within the cabin compartment, performing measurements and experiments, enjoying an amazing view of the Earth and feeling like a real astronaut. In this case, in order to reduce the complexity of the overall SoS, after having performed the parabolic segment and the initial part of the re-entry leg, the vehicle can perform a turn and coming back to the starting airport. In this way, all the required infrastructures can be designed and located in a single place that can also become a touristic attraction itself. This second mission profile can also be considered a safer alternative. Indeed, the dependency from an only on-ground infrastructure allows to design missions that can start from existing airport location and be executed on the sea, on the ocean or on desert areas, in such a way to reduce the flight time over populated areas. Figure 24 shows two different mission profiles designed for touristic purposes. The left-hand side mission profile has been designed during a pre-feasibility study commissioned to ALTEC SpA[1] in 2014 by a group of Malaysian private Stakeholders and carried out by a joint cooperation among ALTEC, TAS-I [2]and Politecnico di Torino. The results of the study have been presented in several congresses and scientific journals (De Vita, 2015), (Fusaro, 2015), (Fusaro, 2015b), (Fusaro, 2017), (Santoro, 2015). In the next chapters, the author will frequently refer to this project and take it as reference because it has been carried out following the steps of the proposed methodology. Please note that in this case, Taiping, a small town located in Malaysia, has been selected to host all the facilities required, the logistic and maintenance buildings, the training centre for passengers and all the infrastructures required to track the vehicle. During the study, a proper area, closed the ocean have been identified to be a possible site for hosting parabolic flight initiatives, minimizing the risk of flying over populated areas. Complementary, the second example proposed in Figure 24 refers to a small feasibility study performed together with Altec and Politecnico di Bari in 2016 (Santoro, 2016) and aimed at evaluating the impact of the envelop of a parabolic mission profile on the Italian territory.

The trend of focusing on this kind of mission profiles is also confirmed by the currently under-development projects of the famous American XCOR (XCOR Aerospace, 2012) and Virgin (Seedhouse,2015) (Figure 25) and the European Swiss Space System (S3), only to make some examples.

[1] ALTEC – Aerospace Logistics Technology Engineering Company – is the Italian centre of excellence for the provision of engineering and logistics services to support operations and utilization of the International Space Station and the development and implementation of planetary exploration missions. ALTEC is a public-private company owned by the major European space company, Thales Alenia Space and the Italian Space Agency, ASI. ALTEC is based in Turin and has liaison offices at NASA and ESA.

[2] Thales Alenia Space, the joint venture between Thales (67%) and Finmeccanica (33%), operates in Italy through Thales Alenia Space Italia, which has 2,300 employees at four plants, in Rome, Turin, L’Aquila and Milan.

As it has been clearly shown in all the pictures of this subsection, both the sketches and the results of trajectory simulations, independently from the specific flight parameters that are directly responsible for the envelope of the profile (i.e. maximum height, maximum horizontal displacement), all the suborbital mission profiles consist of the phases reported in Table 2. The technical specifications, the operative modes and all those characteristics that are more strictly related to the vehicle subsystems or to specific stakeholders' requirements are not considered here but they are in-depth analysed later on. The table summarized the main characteristics of parabolic missions. It is worth to notice that a combination of these phases can always be applied to describe a suborbital parabolic flight, independently from the selected staging or propulsive strategy selected.

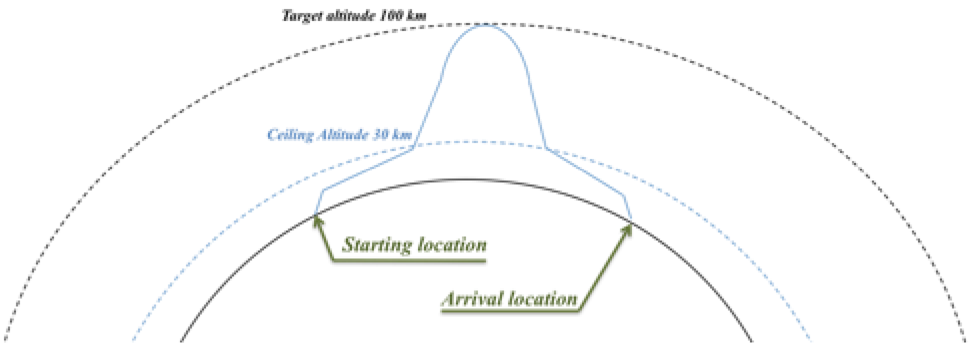


Figure 23: Parabolic Suborbital Mission

Figure 24: Two examples of parabolic flight profiles for touristic missions: on the left, Taiping (Malaysia) and on the right side Grottaglie (Italy).

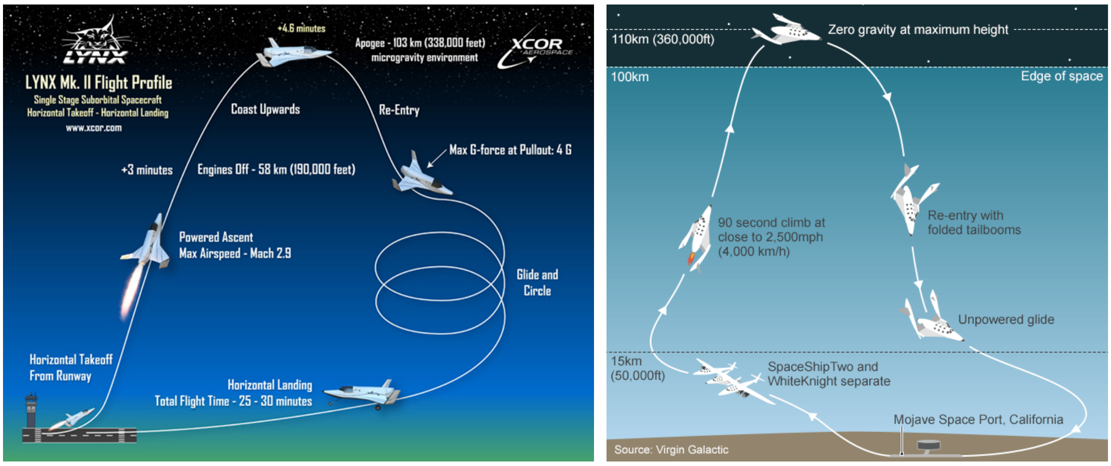


Figure 25: Mission Profiles envisaged by XCOR (left) and Virgin Galactic (right)

Table 2: Description of the phases of a generic suborbital parabolic flight

Mission Phase	Comments
Take-off	Take off strategy directly depends on the propulsive system installed on-board the vehicle and it is often a need or a constraint imposed by the stakeholders. The selection between horizontal and vertical take-off deeply affects the architecture of both the vehicle and the on-ground infrastructure. In case an HOTOL[3] strategy is adopted, the phase can be considered concluded when the aircraft reach a certain altitude, like imposed by the international aviation authority. However, the value of this altitude should be properly addressed by specific studies and properly tailored after safety assessment. In case of a vertical tail sitting strategy, regulatory framework used for man-rated rocket mission can be directly applied. On the contrary, in case of a VTOL[4] strategy in a Harrier-like mode[5], the take-off phase can be considered finished after the transition phase from the pure

[3] HOTOL is an acronym for HOrizontal Take Off and Landing.

[4] VTOL is an acronym for Vertical Take Off and Landing. In this and in the following chapters, this term indicates both tail-sitting and non-tail-sitting startegies.

[5] The adjective Harrier-like mode is used to indicate a VTOL strategy in non-tail-sitting position.

	vertical motion to the horizontal one has been completed. In this case, a proper regulation should suggest the minimum altitude to be reached in pure vertical motion before starting the transition.
1st climb segment	This is the part of climb performed in atmospheric environment, in the aeronautics domain. This is a crucial part of the mission and as it will be explained later on in this document, air-breathing engines are preferable for different reasons (additional margins of manoeuvres in case of deviations from the nominal trajectories, lower acoustic impact, etc…). This phase can be considered concluded after reaching an altitude comprises between 20000 m and 30000 m. This is an indicative altitude that has been considered as reference because it is also the range of altitude for which existing air-breathing engines reach their operative ceiling.
2nd climb segment	This is the segment that allows the vehicle or its final stage to reach the required speed, altitude and attitude to perform the parabolic part of the mission. Considering that this phase requires high level of thrust and it is performed in the outer layers of the atmosphere, rocket engines are currently the only available and tested technology.
3rd climb segment	This phase starts with the rocket burn-out and it is usually performed in an un-powered mode. In some cases, secondary propulsion system, mainly small chemical thrusters, will be employed to correct the attitude or to perform maneuvers[6].
1st Re-entry segment	After reaching the maximum altitude, the vehicle continues its parabolic flight following a ballistic profile. This is a very critical part of the mission because the maneuvers can only be performed by a secondary

[6] During the highest part of the trajectory, depending on particular stakeholders' needs or designers' ideas, there could be the necessity of performing manoeuvres. For sake of clarity, please consider the mission profile hypothesized by Virgin Galactic for it Space Ship II. Just before reaching the top of the parabola, the vehicle is forced to perform a manoeuvre to overturn in order to allow passengers to look the Earth curvature out of the windows that have been installed in the upper part of the fuselage for aero thermodynamic reasons.

	propulsion system because of the quite complete ineffectiveness of flight control surfaces. Moreover, it is usually in this phase that the worst conditions in terms of heat flux and speed are reached.
2nd Re-entry segment	Once the vehicle enters in the lower atmospheric layers, the air-breathing engines can be re-started, or in case of splash-down or soft-landing on un-prepared terrain, flight controls and additional decelerating devices can be used.
Cruise back	It can be considered and regulated as a nominal cruise in the aeronautical domain. Possible high speeds can be reached to shorten the way back.
Descent	The descent can be very different depending the type of mission and on-board equipment. This is one of the most crucial mission phases as far as on-ground tracking concerned because of the very low altitude.
Landing	The landing strategy deeply affects this phase and impacts on the definition of the on-ground infrastructures. From the regulatory standpoint, the considerations performed for the take-off phase can be re-applied.

1.2.2 Orbital Mission Profile

The Orbital Mission Profile (sketched in Figure 26) is not so different from the suborbital parabolic profile analysed in the previous section. The main difference is related to the maximum altitude that is reached at the top of the trajectory and as consequence, the related aerothermodynamics problems. Indeed, during the re-entry phase a higher level of potential energy should be dissipated before entering the lower layers of the atmosphere.

Space Shuttle can be considered the best representative of this kind of mission and the lessons learned after its main accidents are currently considered as guidelines for the design, development and operation of its potential successors. Moreover, an attempt to overcome the Shuttles historical problems (e.g. the capability of providing a cabin escape system or the capability of shortening the turn-around time thanks to a completely reusable thermal protection system) is increasing the public consensus. Especially for governmental organizations, the problem of increasing the public consensus is fundamental for continuing in financing projects. In addition, this is also vital for private commercial initiatives that have often to receive a direct return on ticket selling to continue with the further development steps.

Focusing on Europe, this area of research is also pushed by the need of creating a reusable access to space for European Space Agency (ESA) member States.

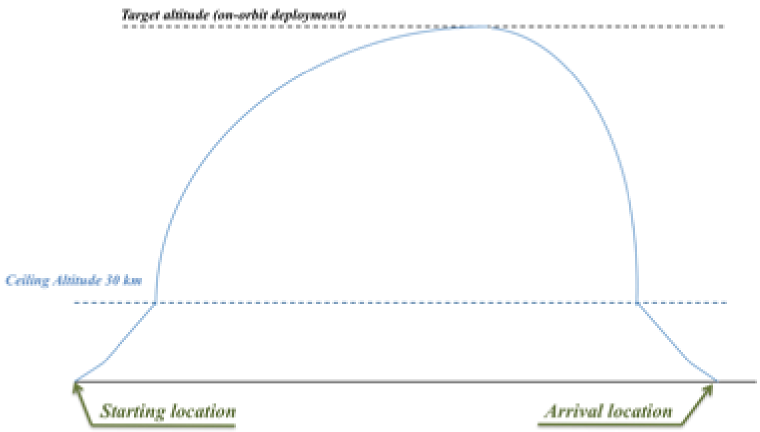


Figure 26: Orbital Mission Profile

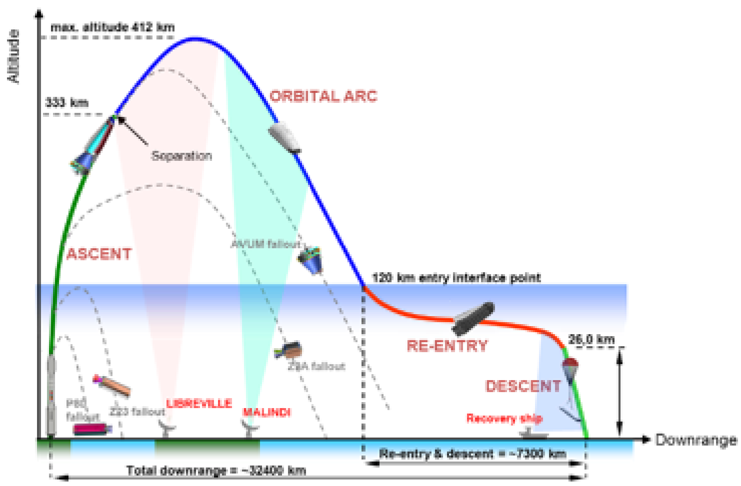


Figure 27: IXV Mission Profile

As it has been clearly demonstrated by the IXV mission, this type of mission profile can be employed for testing the harsh conditions of atmospheric re-entry from orbital conditions. Moreover, this kind of trajectories can also be envisaged for those reusable vehicles aimed at performing orbital deployment incrementing the access to space capabilities. In this context, it is possible to mentioned one of the missions envisaged for the under-development vehicle of the Swiss company S3 and one of the Skylon, the initiative currently under-development in the European possible exploitation (Reaction Engines, 2009). In some cases, the possibility of accommodating and deploying payloads at some pre-defined orbital altitude or concurring to the refurbishment of the International Space Station (ISS) is not the ultimate goal of these vehicle configuration. Indeed, as it has been explained in the Introduction, un-manned missions will act as technology demonstrators, paving the way for the future manned missions.

1.2.3 Point-to-Point Mission Profile

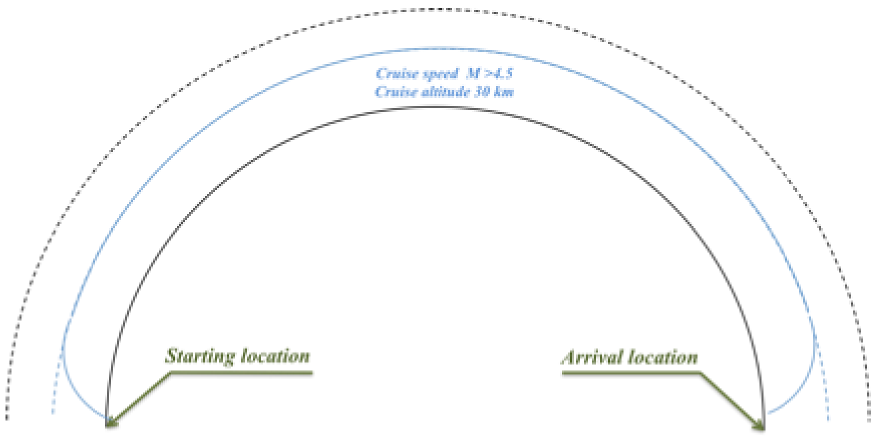


Figure 28: Point-to-Point mission profile

This third mission profile appears to be the most suitable for a Hypersonic Transportation System (Figure 28). In literature, it is also known as Point-to-Point (P2P) mission profile and it consists of a trajectory that is very similar to a common aeronautic long-haul flight profile. However, considering the high-speed, the cruise shall be performed at higher altitude, where the rarefied air can avoid extreme thermal and structural stresses to the vehicle. The exploitation of this new environment, not used in aeronautics and transit environment for space vehicles, forces the engineers to search for innovative technologies especially for the propulsion system.

The possibility of connecting two very far locations on the Earth surface with a single and fast journey, is considered to be the medium and long term future of aviation (both civil and military) and this was clear since the era in which the idea of Concorde born. Moreover, the idea of creating antipodal transportation systems had a great diffusion in the military aeronautical domain, receiving a push during the Cold War, special period during which the main projects aimed at testing technologies for hypersonic have been funded.

But after the pre-mature retirement of Concorde, able to cover the Paris-New York distance in 3 hours and a half flying at a maximum speed of Mach 2,04, the lack of public consensus and several economic crises forced governments to cut off the public funding to several enterprises. Fortunately, although the unfavourable economic conjuncture, the push of the forecast of an exponential growth in the number of passengers, also in long route and especially for business reasons, in the last two decades it is possible to observe a new interest in P2P missions. Examples could be LAPCAT, SpaceLiner, Airbus TBN , Rocketplane, Chinese and the Japanese Spaceplane.

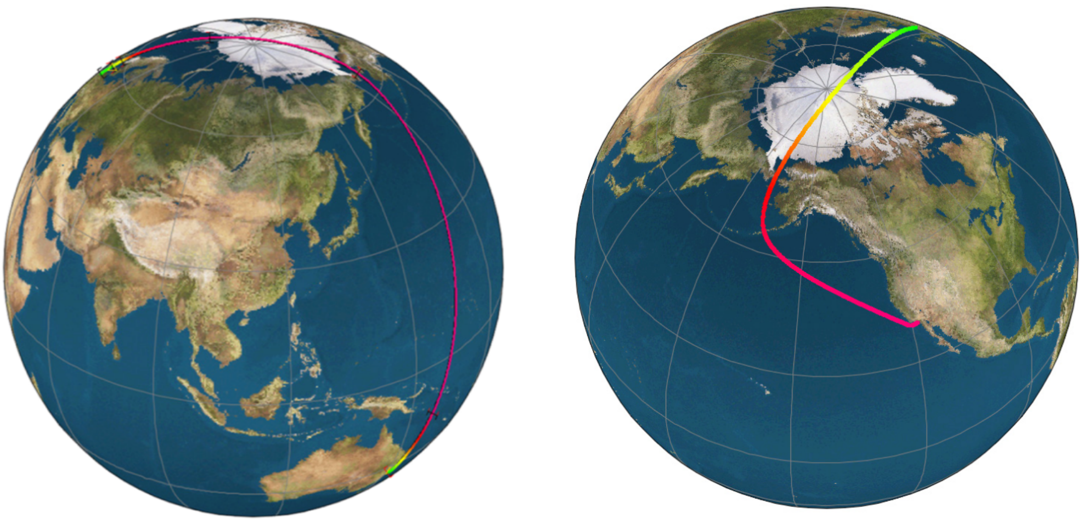


Figure 29: Hypothetic mission of LAPCAT-MR2 (Langener, 2014)

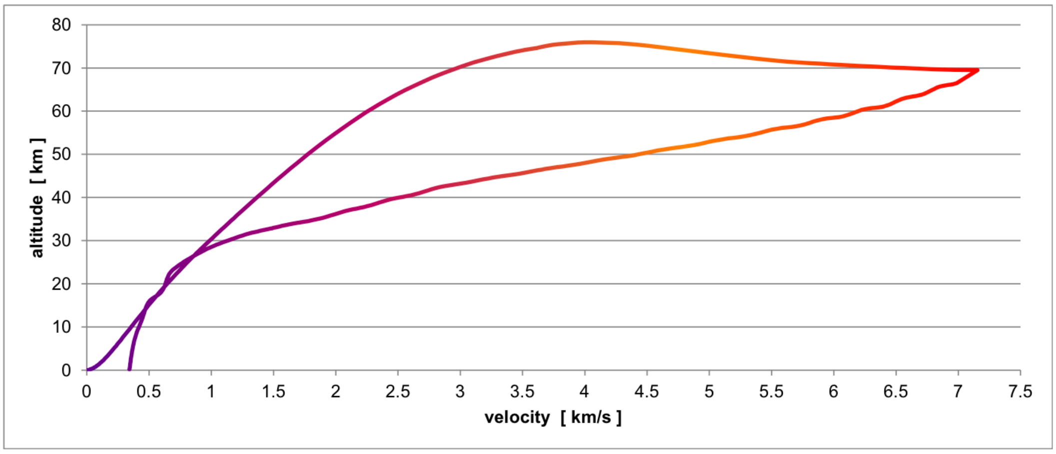


Figure 30: Nominal reference trajectory of Spaceliner 7.1 configuration (Sippel, 2015)

Table 3: Description of the phases of a generic point-to-point hypersonic mission

Mission Phase	Comments
Take-off	Take off strategy directly depends on the propulsive system installed on-board the vehicle and it is often a need or a constraint imposed by the stakeholders. The selection between horizontal and vertical take-off deeply affects the architecture of both the vehicle and the on-ground infrastructure. In case an HOTOL strategy is adopted, the phase can be considered concluded when the aircraft reach a certain altitude, like imposed by the international aviation authority. However, the value of this altitude should be properly addressed by specific studies and properly tailored after safety assessment. In case of a vertical tail sitting strategy, regulatory framework used for man-rated rocket mission can be directly applied. On the contrary, in case of a VTOL strategy in a Harrier-like mode, the take-off phase can be considered finished after the transition phase from the pure vertical motion to the horizontal one has been completed. In this case, a proper regulation should suggest the

	minimum altitude to be reached in pure vertical motion before starting the transition. In the specific case of P2P, it is important to notice that if the vehicle aims at transporting a high number of people, as envisaged by many of the under-development projects, the take-off strategy should be properly evaluated taking into account the easiness of ingress and the comfort to be guarantee to non-trained passengers.
Climb	A single climbing (that could obviously consist of a step-climb) phase can be envisaged for this mission. It can be considered concluded when the vehicle reaches its planned cruise altitude.
Cruise	This phase consists of a forward flight at very high speed and high altitudes. Depending on the locations to be connected, different routes are currently under investigation.
Descent	The descent phase main goal is to decelerate the vehicle up to reach a speed making possible approaching the airport of the arrival location. The descent path is strictly related the vehicle configuration, performances and will be deeply investigated in future with the regulation authority in charge of regulating the airspace in proximity of the airport.
Landing	The landing strategy deeply affects this phase and impacts on the definition of the on-ground infrastructures. From the regulatory standpoint, the considerations performed for the take-off phase can be re-applied.

1.3 Spaceports and Regulations: a new concept for supporting commercial spaceflight initiatives

In general, the term Spaceport designates a site featured with all the technical, infrastructural, operating, safety and the relevant license from the local Authority to allow take-off, landing and ground/flight operations for spaceplanes suitable for suborbital parabolic or point to point flight (Musgrave, 2009)0.

As it is a common saying, in the history of humankind, every great space adventure has begun on the ground. While this seem stating the obvious, mission and spacecraft designers who overlooked with this fact have paid a high price, either in loss or damage to the spacecraft prelaunch or in mission failure or reduction. This means that it is important to consider the on-ground segment, since the beginning of the conceptual design.

This subsection should not only be considered to be a revision of airports with the proper infrastructures required to support the take-off and landing of a spaceplane but to investigate the different aspects that should be taken into account. In particular, special attention should be devoted to the selection of the location of the spaceport, to the sizing of the runway and the different clearance area, the training of the passengers and the medical, logistic and maintenance infrastructures. Moreover, it will clearly appear that a proper set of regulations should be developed in order to guarantee the required safety levels to the on-ground or ground-proximity operations.

1.3.1 Spaceports

1.3.1.1 Selection of suitable spaceport locations

As far as the selection of an appropriate location to build the spaceport is concerned, several aspects can influence the selection process. Depending on the specific requirement of the mission, especially for those transportation systems aimed at reaching the orbit, some latitude and altitude could be preferred. Independently from that, the selection of the spaceport location is mainly guided by the attempt to diminish the level of risk that is related to the presence of inhabitants. It is clear that from the point of view of the safety, the optimal case is to build a spaceport in the middle of nowhere (oceanic or desert regions) limiting the risk of incidents to personnel working on ground infrastructure and also in proximity of the ground footprint of the trajectory. On the opposite, this case is the worst scenario from the point of view of the appeal for tourists. This is the reason why all the major players currently involved in the development and construction of spaceports, are trying to develop attractions in the area closed to the launch pad and to create a global network of connections in order to create a door-to-door one life experience. Then, in a future second phase, with the exploitation of increase TRLs, locations closer to the major international hubs will be selected.

Following the results of a preliminary assessment, the spaceport location shall be such as to allow future activities development and site growth, according to the market predictions. The Spaceport shall be located in scarcely populated areas. In case of horizontal launch operations, the Spaceport shall feature runways 9800 ft long (3000 meters) in both directions. It is also necessary to consider the direction of the prevailing winds and crosswinds in that specific location. For vertical launch operations, the relevant launch platforms shall be placed at a proper distance from the infrastructures for safety reasons.

The problem of selecting a proper location for a spaceport is also related to the capability of performing the mission in segregated areas. Indeed, following the current suggestions of the most prominent rule makers, spaceplane operations shall initially be conducted within segregated air spaces, during the experimental phases unless a sufficient safety confidence level will be reached, although in the future, when suborbital flights will be conducted on a regular basis, it will be necessary to carry out operations in an integrated fashion with the air traffic

The selection of a site as candidate to host a spaceport shall take into account the relevant climatic conditions, with special emphasis to the maximum and minimum annual temperatures, humidity, rain, wind and fog. The possibility of crosswinds could in particular significantly constrain the suborbital operations which are usually conducted with clear skies, in visual flight. In addition, the presence of high altitude wind shear occurrences has also to be considered, in particular between 100000 ft and 200000 ft. These aspects are also significant on a commercial perspective, since the presence of meteorological conditions could limit the visibility and disrupt the overall expected space tourism experience. Taking into account the competitiveness of the initiatives, the environmental impacts resulting from operations, such as water and air pollution, perturbation of ecosystems, noise, and presence of dangerous materials should be properly taken into account. The Spaceport shall comply with the existing laws and regulations, and it is important the support of the general public as well as an adequate information campaign to the local population will be paramount. In particular, in regards to the acoustic impact on the environment, the following aspects shall be evaluated: the spaceplane design and operations optimization, trade-off between costal and inlands areas and cross check with the acoustic environment limitation of the affected area (sonic shockwave boom problem).

1.3.1.2 Spaceports Infrastructures

The spaceport shall support all the operations related to the spaceplane mission and the need of on-ground personnel and passengers. The following table will provide the reader with the major elements infrastructure required to be host in a spaceport.

Moreover, the spaceport shall be equipped with all the infrastructures and facilities needed to prepare the spaceplane for flight, along with the payload if applicable and the flight participants, passengers and crew

In particular, the Spaceport shall be equipped with the infrastructures and capabilities to the preparation and execution of the vehicle pre-flight test and checkouts along with the relevant subsystems and equipment to ensure functional and interface checks and integration activities, the Spaceport shall feature the proper capabilities to perform proper prelaunch integration and configuration. For example, in the case of the Virgin Galactic White Knight 2 and Space Ship 2 integration activities, or vertical take-off multistage final integration and test.

The Spaceport shall be equipped with all the needed infrastructures that allow preventive and corrective maintenance on the vehicle and with a proper system to manage the spare parts and relevant procedures.

The Spaceport shall be equipped with all the infrastructures needed to store dangerous materials like propellant and explosives in accordance to the specific spaceplanes. Such infrastructures shall have to be built and handled in accordance with the national regulations and Safety. Such infrastructures shall also feature temperature and humidity control capabilities and lightning protection. Specific containment areas shall be envisaged to mitigate risks resulting from emergencies such as explosions, fire, overpressure and toxic materials release. In accordance with the Safety requirements, the maximum amount of dangerous materials is a function of the distance from the operating areas of the Spaceport and mostly from populated areas and roads (FAA, 1999). Basing on this, this may be a major constraint when evaluating already existing sites, because such safety distances may not be compatible with the layout of already existing sites. The Spaceport shall be equipped with a proper radio communication infrastructure, operating both on aeronautic used frequencies and internal offline ones to allow proper execution of ground and flight operations

The Spaceport shall include planning capabilities to support both flight and ground operations. This includes also resources allocation and the necessary resources in the adequate times.

The Spaceport shall include proper training facilities for flight crew, flight participants and flight passengers for space tourism.

The Spaceport shall feature the proper capabilities aimed at the coordination and support of the launch operations and the initial phases of the flight, in the vicinity of the launch site as follows:

The Spaceport shall be equipped with the proper infrastructures to allow monitoring of the ascent phase and stages separation, when applicable. The ascent trajectory shall be accurately planned with respect to interference with active airspace, and debris fallout areas in case of system fragmentation following emergency or failure.

The Spaceport shall be able to support the spaceflight mission operations and in particular shall be able to handle emergency situations that may result in e vehicle return to the launch site. Specific coordination with the various ATC of the affected areas is required.

The Spaceport shall be able to handle emergency situations and people rescuing in coordination with the proper Organizations

The Spaceport shall support the spaceplane descent, final approach and landing operations in coordination with the with the various ATC of the affected areas

The Spaceport shall feature the proper infrastructures to execute all the planned post landing operations; these include crew, flight participants, passenger disembarking, spaceplane saving, post flight check of the spaceplane system and equipment, vehicle refurbishment and reconfiguration to the next mission. A proper data archiving system maintenance record and configuration control systems shall be in place.

The Spaceport suitable for Space Tourism shall include all services to passengers that will enhance their spaceflight experience. These services are included in the package most of the Operators are planning to offer to their customers. Such services include training, medical, entertainment, public observation areas.

Safety plays a paramount role in Spaceports evaluation, with the purpose of identifying, analyzing, remove or reduce all risks to the general public, the site personnel and the relevant infrastructures. The risk reference value associate with the Spaceport activities is assumed to be less than 30×10^{-6} casualties (FAA, 2009).

The Spaceport Operator shall organize and manage all the Spaceports Safety aspects by setting up in particular the following activities:

- Define the Safety Organization and the Safety Manager
- Prepare the Safety Management Plan
- Manage, approve, coordinate through the Safety Team the implementation of the Spaceport Safety Requirements.
- Be actively involved and participate via the Safety Review Board in any change or update affecting Spaceport functions and operations,
- Establish and keep current an initial Spaceport Safety Training Program.
- Define, analyse and manage the Safety aspects for the Spaceplanes operating in the Spaceport.
- Make sure that all the Safety functions have properly been implemented for the Spaceport Operations.
- Ensure that the Spaceport personnel and visitors have properly been informed about the possible risks associated with the Spaceport Operations and the actions to be initiated in case of emergency.
- Participate in the Certificate of Flight Readiness Process for all Safety aspects
- Define an Emergency Response Plans (ERP) and ensure proper coordination with the designated counterparts to prevent or mitigate the exposure of the general public and personnel to the risks related with the Spaceport Operations
- Ensure that the designated Safety Personnel has the proper qualification and training to fulfil the Safety requirements and procedures

- Ensure that the Spaceport Safety Team is always involved in the relevant operations, pre-flight test and critical operations like the propellant loads.
- Ensure proper coordination with the various authorities like Civil and Military Aviation, ATC for all Safety aspects related to the planned operations.
- Organize and participate in periodic simulations relevant to the Spaceport Safety Processes.
- Participate in the real time Spaceplane trajectory and in the relevant systems and equipment real time monitoring in relevance to the Safety aspects.
- Participate in the decision process to determine a possible flight interruption/abort in case of Safety criteria violations both for the Spaceport perimeter and for the surrounding area.

When setting up a Spaceport a proper Safety Management System is a fundamental step. Such a system shall be prepared 'ad hoc' by the Spaceport Operator for the specific activities to be carried out. Purpose of the Safety Management System is to put in place a proper organization for the management of all the Safety aspects. The Safety Management System shall properly be documented in a dedicated plan for the specific Spaceport. When evaluating existing sites as possible candidates to become Spaceports, the approach to be followed is the identification of all the hazards associated with all the Spaceport Departments and all the Spaceports/Personnel activities; once the hazards have been identified, they will be documented in a proper hazard log and a proper analysis has to be carried out of to derive the consequent risks and identify the proper control actions to mitigate the risks to the maximum extent, such to prevent their possible evolution in accidents. The described process might eventually result in the need of implementing specific modifications or upgrades to the existing infrastructures.

The Safety Management System represents the prime reference for the daily implementation of the Safety approaches defined for the specific Spaceport.

Table 4: Infrastructure for spaceports

Class of Infrastructures	Features Description

Operative Infrastructure: all those infrastructures aimed at supporting the operations of the spaceplane	Runway Launch pad Tower Railhead
Logistical and maintenance infrastructures	Fuel storage Fuel handling Chemical analysis facilities Vehicle checkout Spacecraft storage facilities Engineering/mission management offices Radars Telemetry data Engine test Material Testing facility
Additional Infrastructure: all those infrastructures that are not strictly related to the operations but they are vital for the spaceport	Road access Medical facilities Training facilities Simulators Space academy
Touristic Infrastructure	Hotel

	Restaurant
	Thematic Entertainment Park
	Thematic Shops
	Family facilities/residential
	Family facility facilities/entertainment

1.3.1.3 Operative and under development spaceports

Spaceport in US

The first spaceport built is Spaceport America. Thanks to the fact that is home to Sir. Richard Brandson' Virgin Galactic, the world's first commercial spaceline, the project was able to attract a very high number of fundings. The New Mexico Spaceport Authority (NMSA) is responsible for designing, building and operating this spaceport that consists of an airfield, launch pads, terminal, hangars and all the required roadway connections, allowing both HOTOL and VTOL spaceplane operations. In addition, it can provide support for astronauts training activities, but can also entertain accompanying person in case of touristic missions. Considering the Spaceport America Business plan (Spaceport America, 2016) for the period 2016-2020, the developers and operators of this infrastructure should pursue these three main objectives:

- First and foremost, the primary strategic goal for Spaceport America is to deliver efficient and effective services to all customers.
- The second major strategic goal for Spaceport America is to drive local job creation and inject the economy with greater demand for goods, services and skilled workforce.
- The third major strategic goal is to inspire our guests, particularly the next generation.

Unless there are lots of initiatives all around the world, Spaceport America is the only one currently built from scratch and properly designed to offer a regular flight service and its relevance has been understood by the New Mexico citizens that voted for a sales tax aimed at supporting the construction.

The location of this spaceport, in the middle of a desertic area at more than 50 km from the closest city, is the optimal solution as far as the safety is concerned. Unfortunately, being in the middle of nowhere is a limiting factor from the point of view of the touristic appealing. For this reason, a connection network should be envisaged in order to offer a well-structured door-to-door experience.

In order to support the economy of the spaceport and to support the commercial flight activities, an integrated touristic experience is proposed. The current Spaceport America tour is attracting approximately 3,000 visitors per year. The new expanded Spaceport America Experience is expected to attract thousands more the first year and grow exponentially each year with the onset of the commercial flight activities. The tour begins in the nearby town of Truth or Consequences in the historic downtown hot springs district in a unique 1936 adobe building that has been fitted-out with educational and fun space related exhibits. The Spaceport America shuttle bus departs from there for a 45 minute guided journey to the Spaceport America site. Guests will be entertained on the bus by several original videos showcasing the history of the region from the Paleo-Indians; the trading route from Mexico to Santa Fe in the 1500's; the Atchison, Topeka and Santa Fe railway in the 1800's; and the beginning of the first space age with pioneers like Goddard and von Braun. Guests will then experience the Gateway Gallery in the award winning Virgin Galactic Gateway to Space building. The Gallery has interactive kiosks, a G-Shock simulator, original videos and much more. Guests are then treated to a tour of the Spaceport Operations Center, a ride down the 12,000 ft spaceway (runway) and a photo-stop outside the Gateway to Space building.

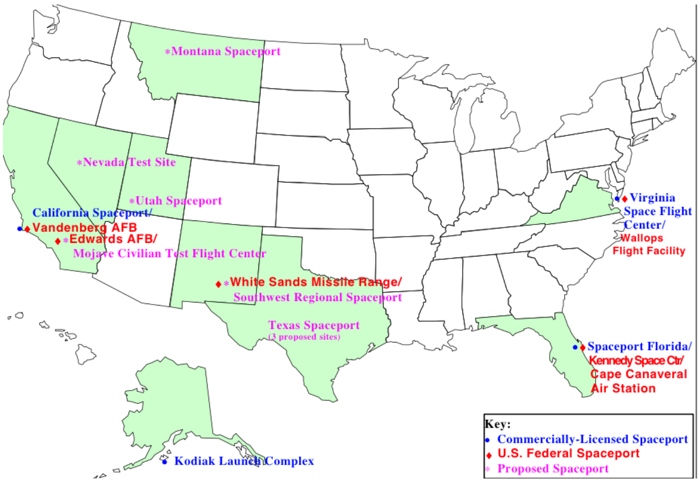


Figure 31: Location of spaceports in US [2000RLV]

Although the most important initiative is Spaceport America, other activities are currently under development in the US like the Mojave and the Florida Spaceports.

Figure 32:Spaceport America

FY15 Investment		Return on Investment(ROI)/Economic Stimulus	
State of NM Annual Investment	$463,000	Spaceport Funding for STEM-education	$1,900,000 [1]
		Taxes Paid on Operations	$160,000
		Taxes Paid on Construction	$70,000 [2]
		Taxes Paid on SA Events	$6,000 [3]
		Direct Spend by SA Events	$43,000 [4]
		Direct Spend by Tourism	$177,000 [5]
		Direct Spend by Tenants	$1,000,000 [6]
		Taxes Paid by Tenants	$70,000 [7]
		Local Space Conference Economic Impact	$1,293,000 [8]
		SA Employment Direct Impact	$861,000
		SA Employment Indirect Impact	$258,000 [9]
		SA Contractor Employment Direct	$900,000
		SA Contractor Employment Indirect	$270,000 [9]
		SA Tenant Employment Direct	$1,960,000
		SA Tenant Employment Indirect	$588,000 [9]
Total	**$463,000**	**Total**	**$9,556,000**

[1] Per NM Tax & Revenue Statistics
[2] Assumes $1M annual construction spend
[3] Based on 4 year average of $85K per year in event spending
[4] Assumes 0.5X multiplier of onsite spending spent
[5] Assumes 3000 visitors in 2015
[6] Assumes $1M average offsite local spend per year
[7] Based on Direct Spend
[8] Per NM Space Grant Consortium
[9] Assumes 0.3X multiplier of local salary impact

Figure 33: Spaceport America expected Return on Investments (ROI)

Table 5:US spaceports

Spaceport	Location	Spaceport Owner/Operator	Launch Infrastructure at Site	Current Development Status
Commercially Licensed Spaceports				
California Spaceport	Lompoc, California	Spaceport Systems International, L.P.	Existing launch pads, runways, payload processing facilities, telemetry and tracking equipment.	The site is still undergoing upgrades. Currently in place are the concrete flame ducts, communication, electrical, and water infrastructure.
Kodiak Launch Complex	Kodiak Island, Alaska	Alaska Aerospace Development Corporation	Limited infrastructure at this time.	Construction for the launch control and management center, the payload processing facility and the integration and processing facility is in progress.
Spaceport Florida	Cocoa Beach, Florida	Spaceport Florida Authority (SFA)	Two launch complexes including pads and a control center, a small payload preparation facility and an RLV support facility.	SFA has invested over $200 million to upgrade LC 46 and 20, build an RLV support complex adjacent to the Shuttle landing facilities, and develop a new space operation support complex .
Virginia Flight Test Center	Wallops Island, Virginia	Virginia Commercial Space Flight Authority	Launch pad and service tower, payload processing facility, downrange tracking facility.	Pad 0-B was completed in December 1998. VSFC obtained a commercial license from the FAA in 1997.
Federal Spaceports				
Cape Canaveral Air Station/ Kennedy Space Center	Cocoa Beach, Florida	USAF, NASA, Florida Spaceport Authority	Telemetry and tracking facilities, jet and shuttle capable runways, launch pads, hangar, vertical processing facilities and assembly building.	RLV spaceport and processing facilities to be completed in early 2000.
Edwards AFB	Mojave, California	USAF	Telemetry and tracking facilities, jet and shuttle capable runways, X-33 launch pad, operations control center, movable hangar, fuel tanks, water tower.	Site is operational.
Vandenberg AFB	Lompoc, California	USAF	Launch pads, vehicle assembly and processing buildings, payload processing facilities, telemetry and tracking facilities, control center, engineering office space, shuttle-capable runway.	VAFB has started negotiations with several commercial companies. Existing infrastructure is operational. Upgrades may or may not be required depending on vehicle requirements.
Wallops Flight Facility	Wallops Island, Virginia	NASA	Launch pads, blockhouses and processing facilities.	Wallops Flight Facility has not supported any orbital flights since the failure of Conestoga in 1995. NASA is committed to maintain the existing infrastructure.
White Sands Missile Range	White Sands, New Mexico	US Army	Telemetry and tracking facilities, 4.5 km runway. Engine and propulsion testing facilities.	NASA Flight test center is operational. RLV-specific upgrades will probably be required.
Proposed Spaceports				
Montana Spaceport	Great Falls, Montana	Montana Space Development Authority	No infrastructure at this time.	Montana Spaceport is primarily seeking RLV business. The Montana Space Development Authority is in the process of obtaining a commercial spaceport license for the Great Falls site.
Nevada Test Site	Nye County, Nevada	Department of Energy/Nevada Test Site Development Corporation (NTSDC)	No launch infrastructure at this time. Power and basic facilities available.	NTSDC has issued a sub-permit allowing Kistler to operate a launch and recovery operation. NTSDC is actively promoting the site as a spaceport for both RLVs and conventional launchers.
Southwest Regional Spaceport	White Sands, New Mexico	New Mexico Office of Space Commercialization	No infrastructure at this time.	Plans for this site include a Spaceport central control facility, an airfield, a maintenance and integration facility, a launch and recovery complex, a flight operation control center, and a cryogenic plant.
Texas Spaceport	TBD	State of Texas Spaceport Authority	No infrastructure at this time.	The final Texas Spaceport site(s) has not been selected yet.
Utah Spaceport	Wah Wah Valley, Utah	Utah Spaceport Corporation	No infrastructure at this time.	Plans for the proposed Utah Spaceport include a central administrative control facility, an airfield, a maintenance and integration facility for both payloads and craft, launch pads, a flight operation control center, and a propellant storage facility.
Woomera Rocket Range	Woomera, Australia	Woomera Rocket Range/Kistler Woomera	Outdated launch site infrastructure, assembly building, and tracking facility.	Kistler is starting work on new facilities in January of 2000. The Australian government has passed congressional acts to attract aerospace business to the Woomera area.
Mojave Civilian Test Flight Center	Mojave, California	Mojave Airport Authority	Air control tower, runway, rotor test stand, engineering facilities, high bay building.	The infrastructure in place is part of a $5.5 million project.

Spaceport Malaysia

Considering the Eastern Countries, the most active in the field of increasing access to space with a special attention to hypersonic flight and space tourism is Malaysia. According to (Zakaria, 2013), the project is articulated in several phases. In particular, during the first phase of the program, well-known space industries will be involved in the operations, envisaging activities like commercial zero gravity flights, small satellite launch using commercial jet planes and commercial sub-orbital spaceflights. Then, in the later phases, the experience accumulated during the first phases, routine commercial point-to-point suborbital and orbital flights will be guaranteed to address the growing space tourism demands and activities. Spaceport Malaysia envisages to become the reference Asia′s space hub for scientific research, astronautic flight and micro satellite launching. A part from containing all the basic elements of an advanced spaceport, the planning and design of Spaceport Malaysia is based on four important factors: human factors, machine interface, environmental compatibility and safety. These unique combinations are the core design principles guaranteeing technological advancement, climate factors and cultural development in the spaceport design.

Malaysia government is also very interested in finding cooperation with foreigner space agencies and industries. In particular, the July 29th, 2015, at the headquarters of the Italian Space Agency (ASI) in Rome, the President Roberto Battiston received a delegation of Malaysia, led by Abu Bakar Bin Mohamad Diah, Deputy Minister for Science, Technology and Innovation. Deputy Minister was accompanied by a delegation of industry and society Space Ventures that on July 28th, in Turin, has signed an agreement with ALTEC, for the study of issues on orbital flights.

Figure 34: Spaceport Malaysia (Top View)

Figure 35: Spaceport Malaysia (Artistic View)

Spaceport in Europe

Kiruna may seem an unlikely place to build Europe's first commercial spaceport. Its 67,86° latitude means it is 150 km above the Artic Cycle and close to 900 km north of Stockholm. In addition to its extreme grid reference, Kiruna has several disadvantages: vast expense forests, no sunlight for weeks and very harsh temperatures. In 2007 the government announced an "agreement of understanding" with Virgin Galactic to make Kiruna the company's first launch site outside the US. Please consider that additional initiatives are currently under development in other European Countries, in particular in England, Spain but also in Italy as it has been mentioned in the previous sections.

1.3.2 Regulations and Licensing

The Spaceport shall obtain the proper license by the relevant Civil Aviation Authority, to operate the specific suborbital vehicle basing upon the definition of a specific regulatory framework.

The FAA provides different useful licenses for the RLVs, as the license for launch or reentry vehicles and the license for launch or reentry sites. The process needed to obtain one of the FAA licenses starts with a common preliminary phase called Pre-Application Consultation, thanks to which the applicant can became familiar with the regulatory framework. When he is ready, the applicant can submit the formal application to the FAA, following the guidelines provided by the 14 Code of Federal Regulation (CFR) chapter III. The general flow scheme of the FAA licensing approach is shown in Fig. 3.

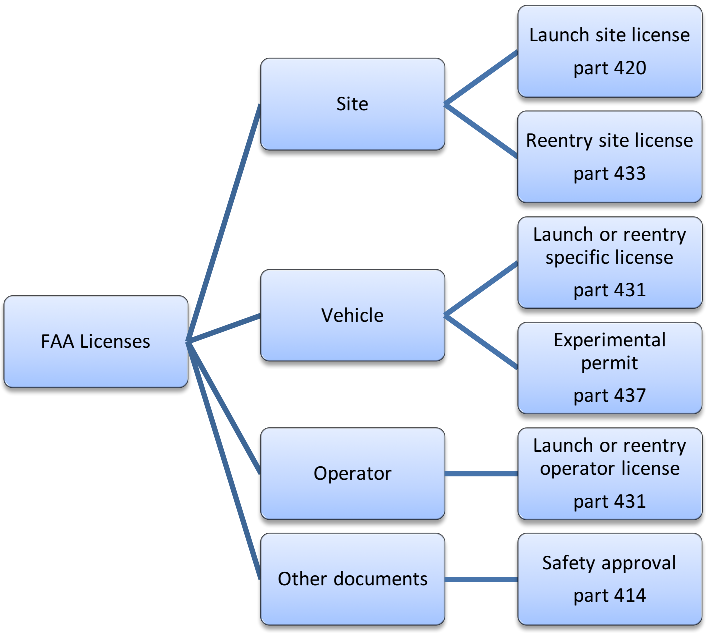


Figure 36: FAA licences for RLVs

The licensing process for a launch site is based on the 14 CFR chapter III part 420, and it consists of different steps that evaluates any possible issues of the license affecting the U.S. national security and policy, the capacity to conduct operations in compliance with the safety regulations, a constant control post – license in order to verify that the licensee is operating in accordance with his application, and an environmental assessment of the launch site activities. The license enabling the licensee to offer its launch site to a launch operator for each launch point for the type and any weight class of launch vehicle identified in the license application and upon which the licensing determination is based.

The reentry site license is based on the part 433, and it authorizes the licensee to offer use of the site to support reentry of a reentry vehicle for which the three-sigma footprint of the vehicle upon reentry is wholly contained within the site. The process in order to issue the license is the same provided by part 420, without the phase of compliance monitoring.

The FAA launch or reentry specific license, based on the part 431, authorizes to conduct one or more launches or reentries having the same operational parameters of one type of launch or reentry vehicle operating at one approved launch or reentry site. The authorization terminates when all authorized launches or reentries are completed by the licensee or at the expiration date stated in the license. Besides the phases of the launch site licensing process, the launch or reentry vehicle licensing process also requires a payload review, a compliance with part 460 if there are crew and/or space flight participants on board the vehicle, and a determination of financial responsibility (part 440), which is based on the determination of the Maximum Probable Loss (MPL), a dollar value depending on an analysis and assessment of the maximum monetary losses likely to be incurred by the government and third party personnel and property in the event of mishap. According to the FAA, the federal government takes over the liability to third party not involved in the flight and put a limit to the liability of operators to third party damages. The federal level indemnification in case of accident is then complemented by indemnification regimes at state level.

The launch or reentry operator license is based on the part 431 too, and it authorizes to launch and reenter, or otherwise land, any of a designated family of RLVs within authorized parameters, including launch sites and trajectories, transporting specified classes of payloads to any reentry sites or other location designated in the license. The procedural steps to obtain the operator license are the same required for the launch or reentry specific license, with the only difference that a launch or reentry specific license licenses only a specific launch or reentry activity.

The FAA provides also other two documents called experimental permit and safety approval.

Experimental permits, stated by part 437, are authorizations dedicated to the development of Reusable Suborbital Rockets (RSR), issued to test new design concepts, new equipment, new operating techniques, crew training before obtaining a license for a launch or reentry. The permit allows to conduct an unlimited number of launches and re-entries for a particular suborbital rocket design, without carrying any property or human being for compensation or hire. Overall the issuance of an experimental permit requires an analysis of the financial responsibility, a safety approval, and an environmental assessment.

A safety approval, stated by part 414, contains a determination that one or more safety elements will not jeopardize public health and safety or safety of property. Safety elements include a launch vehicle, a re-entry vehicle, a safety system, process, service, or any identified component thereof, and qualified and trained personnel performing a process or function related to licensed launch activities. It may be issued independent of a license and it could be also used as an instrument in the decision-making process, having the chance to evaluate different safety elements before applying for a licensing process, such us the choice of the best landing site for a re-entry vehicle. Thanks to the Memorandum of Cooperation between ENAC and FAA signed in 2014, it is clear the chance to use the FAA regulatory framework as an input to elaborate a national framework, without incurring in some of the problems connected with the ITAR.

Consequently, it is fundamental to review the FAA processes under the guidance of a national aviation authority like ENAC (ENAC, 2016), trying to understand the uncertainty elements, which will need to be analysed in a more detailed way in order to comprehend if they have to be changed or removed to align the U.S. approach to the Italian context. In fact, while it is assumed to adopt the same safety level of the FAA (30 casualties10^{-6} missions), elements like the agreements with the local FAA Air Traffic Control and the U.S. Coast Guard to protect public health and safety (stated by part 420, 431, and 437) should be enriched with specific agreements between the operator and all the subjects that will take part in the operations, like the air force, local emergency authorities, etc. About the financial responsibility and allocation of risk, it is clear how it will be necessary to define national financial and insurance requirements for a licensee or a permittee, and the amount of money for liability of covered claims, modelling a regulatory framework based on both the U.S. and the current European and national laws. The informed consent with which the space flight participants are informed about the risks of the launch or re-entry represents another key element and it should be analysed to understand its actual legal value in the national context. Eventually, it will occur to adapt the U.S. environmental program to the European and national rules. However, in the short period some of these critical features could be tackled introducing a bilateral agreement between U.S. and Italy based on a wet – lease approach, to define responsibilities and procedures allowing U.S. personnel to conduct suborbital operations within Italy.

References

Allaway, H. (1979). The Space Shuttle At Work.

Associate Administrator for Commercial Space Transportation, (2000) Reusable Launch Vehicle Programs & Concepts, with special section on Spaceports

Boeing (1999) X-37 Demonstrator to Test Future Launch Technologies in Orbit and Re-entry Environments," *Boeing Press Package*, July 14, 1999.

Bonetti, D. IXV Mission Analysis and Flight Mechanics: from design to postflight. *23rd Conference of the Italian Association of Aeronautics and Astronautics, AIDAA2015*, Politecnico di Torino, Italy. 2015.

Chang, Kenneth (2014). 25 Years Ago, NASA Envisioned Its Own 'Orient Express'. The New York Times. Retrieved 21 October 2014.

De Vita, Francesco, et al. (2015) Assessment of Hypersonic Flights Operation Scenarios: analysis of launch and reentry trajectories, and derived top level vehicle system and support infrastructure concepts and requirements. *20th AIAA International Space Planes and Hypersonic Systems and Technologies Conference*., Glasgow (UK).

ENAC (2016), A Regulatory Policy for the Prospective Commercial Space Transportation Certification and Operations in Italy (2016)

FAA (1999), Commercial Space Transportation , Federal Aviation Administration, Department Of Transportation, CFR 14, Part §420

Fusaro R. et al. (2015) Conceptual design and operations of a crewed reusable space transportation system." *IAC Congress*, Jerusalem (Israel).

Fusaro R. et al. (2015b) Conceptual design of a VTOL spacecraft aimed at parabolic flights. At *AIDAA Congress*, Turin (Italy), 2015.

Fusaro R., et al. (2017) "Conceptual design of a crewed reusable space transportation system aimed at parabolic flights: stakeholder analysis, mission concept selection and spacecraft architecture definition. *CEAS Space Journal* 131, doi :10.1007/s12567-016-0131-7. 2016

General Accounting Office (1991). Aerospace Plane Technology: research and Development Efforts in Europe, U.S. GAO/NSIAD-91-194, July 1991, p.124.

Guthrie, J. (1988) Spaceplanes and the rise of 'Ultra Tech'. *Hypersonic Flight in the 21st Century.* Vol. 1.

Hannigan, R. J. (1994) Spaceflight in the Era of Aero-space Planes. Krieger Publishing Company.

Hechler, K. (1982). The endless space frontier: a history of the House Committee on Science and Astronautics, 1959-1978. Vol. 4. *American Astronautical Society*, 1982.

Koelle, D. E., and Heribert Kuczera (1990). SÄNGER Space Transportation System-Progress Report 1990. IAF, 1990.

Kuczera, H., et al., (2011) Reusable Space Transportation Systems. UK: *Praxis Publishing*, 2011. ISBN: 978-3-540-89180-2 1 .

Kuczera, Heribert, and Helmuth Hauck (1992). The German Hypersonics Technology Programme—Status Report 1992. *43rd Congress of the International Astronautical Federation*. 1992.

Langener T., et al. (2014) Trajectory simulation and optimization of the LAPCAT-MR2 hypersonic cruiser concept, *29th ICAS Congress*, St. Petersburg (Russia).

Larson, W. J. et al. (1999). Human spaceflight: mission analysis and design. *McGraw-Hill Companies.*

Musgrave, G. E. et al. (2009). Safety design for space systems. *Butterworth-Heinemann*, 2009.

Parkinson, R. C. (1990). A total system approach towards the design of future cost-effective launch systems. *IAA, Symposium on Space Systems Cost Estimation Methodologies and Applications*, San Diego, CA.

Raymer, Daniel P., and Mitchell Burnside Clapp. (2002) Pioneer rocketplane conceptual design study. Journal of aircraft 39.3 (2002): 507-511.

Reaction Engines (2009), SKYLON Users' Manual – Rev. 1.1, Reaction Engines Ltd (UK).

Santoro F., et al. (2013), The Italian spacegate: study and innovative approaches to future generation transportation based on high altitude flight, *64th International Astronautical Conference*, Beijing China 2013,

Santoro F., et al. (2015) Approaches to the development of commercial spaceport and associated ground segment driven by specific spaceplane vehicle and mission operation requirements. *IAC Congress*, Jerusalem (Israel).

Santoro F., et al. (2016) Advances in Italian spaceports identification as infrastructures for suborbital flight activities and mission profiles. *Proceedings of the International Astronautical Congress, IAC*, Guadalajara (Mexico).

Seedhouse, E. (2014). Tourists in space: A practical guide. 2nd Edition, *Springer Science & Business Media*.

Seedhouse, Erik (2015). Virgin Galactic: The First Ten Years. *Springer*, 2015.

Sippel et al. (2015). SpaceLiner Technical Progress and Mission Definition, *20th AIAA International Space Planes and Hypersonic Systems and Technologies Conference*, Glasgow (UK) 2015

Spaceport America (2016), Spaceport America Business Plan, bringing the future to the present 2016-2020 (2016)

Steelant J. et al. (2015). Achievements Obtained for Sustained Hypersonic Flight within the LAPCAT-II Project, *20th AIAA International Space Planes and Hypersonic Systems and Technologies Conference*, Glasgow (UK) 2015

Steelant J. (2009). Sustained Hypersonic Flight in Europe: Technology Drivers for LAPCAT II, *16th AIAA/DLR/DGLR International Space Planes and Hypersonic System Technologies Conference*. Bremen, Germany: AIAA 2009-7240, 2009.

Steelant J. (2008). Achievements obtained for sustained hypersonic flight within the LAPCAT project , *15th AIAA International Space Planes and Hypersonic Systems and Technologies Conference*. Dayton, OH: AIAA 2008- 2578, 2008.

Tauri Group (2012) "A 10-Year Forecast of Market Demand." The Tauri Group under contract with the Federal Aviation Administration and Space Florida (2012).

Tumino, G. at al (2008). The IXV Project. The ESA Re-Entry System And Technolologies Demonstrator Paving the Way to European Autonomous Space Transportation and Exploration Endeavours. *59th International Astronautical Congress*, Glasgow, IAC-08-D2.6.1, 2008

Tumino, G., L. Basile, and E. Angelino. The IXV Project from Design to Development. *IAC-09-D2*. 6.03, 2009.

Weiland, C. (2014) Aerodynamic Data of Space Vehicles. *Springer Science & Business Media*.

Wilson, Andrew (1986). Space shuttle story. *Crescent*, p.14.

XCOR Aerospace (2012), XCOR Lynx Payload User's Guide, Versio3b, California (USA).

Zakaria R. N., et al. (2013). Human, Machine, Nature and Safety Factors in the Design and Architecture of Spaceflight Terminal at Spaceport Malaysia. Safety is Not an Option, *Proceedings of the 6th IAASS Conference*. Vol. 715. 2013.

Chapter 2

Integrated Design Methodology and a support tool-chain

2.1 The need of an integrated design methodology and the importance of a support tool – chain

Dealing with complex and very innovative aerospace systems implies extension of time required for the design, development and production as well as the increase of risks and costs associated to the overall life cycle. In particular, this thesis focuses on the design phase and this section is devoted to the identification of the major reasons why an integrated multidisciplinary conceptual design methodology may be envisaged, especially for hypersonic vehicles.

First of all, the application of an integrated design methodology since the very beginning of the design process will allow:

- To shorten the design activities reducing the number of design iterations.
- To limit the economic risk related to unappropriated design solutions, increasing the confidence level of the high level estimations and postponing as much as possible the freezing of the baseline.

Unfortunately, the design process of highly innovative transportation systems is hampered by several different elements of complexity.

- The high level of physical integration that forces to develop a methodology able to deal with different levels of detail, tracing the impact of systems and subsystems integration back to vehicle levels. In this context, a fundamental role is played by requirements management activities.
- The high level of innovation of the product should be properly managed because it could deeply affect strategic decisions, implying consistent changes to schedule and budget. In aerospace, Technology Readiness Level (TRL) is the most widely used index to define the level of maturity of a technology. However, considering the high level of innovation (and thus, low TRL) and the high level of integration of the products we are dealing with, other Figures of Merits have been included within the methodology: the System Readiness Level (SRL) and the Integration Readiness Level (IRL). Thus, the integration of a Technology assessment process should be considered since the beginning (**Errore. L'origine riferimento non è stata trovata.**).

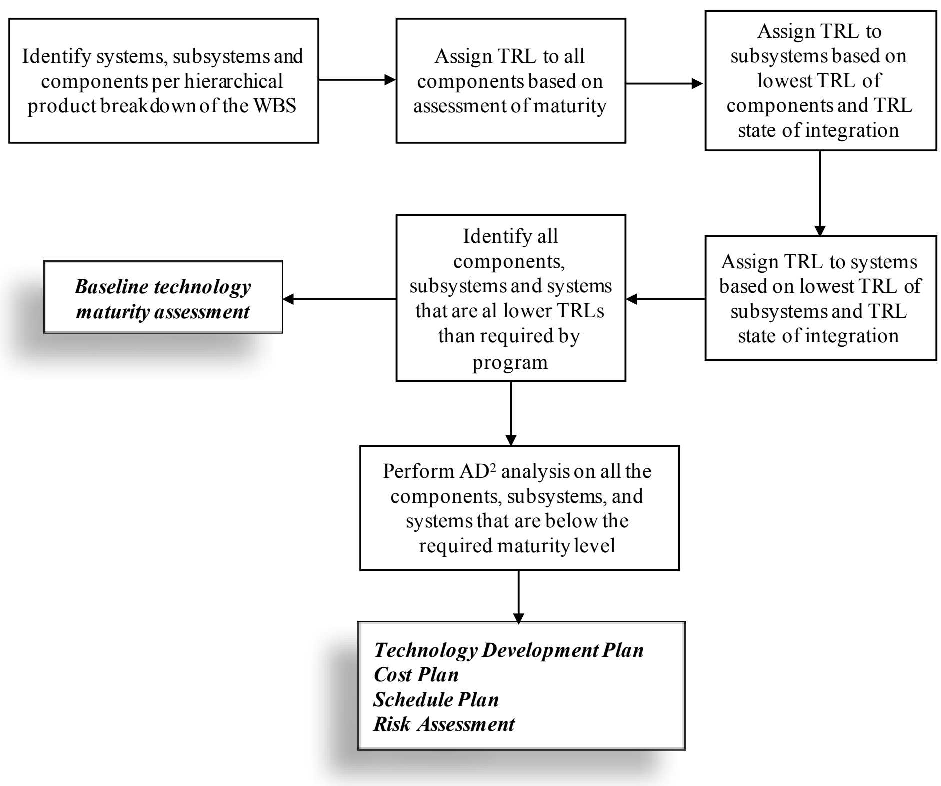


Figure 37: NASA Technology assessment process (NASA, 2007)

2.1.2 TRL, SRL and IRL

Before entering in the detail of the suggested design methodology, this section introduces some definitions of useful parameters connected to the readiness level of systems and technologies. Considering what has been stated by Mankins (Mankins, 2009), the concept of Technology Readiness Level was introduced by NASA in the mid 1970s, with the aim of creating a discipline-independent FoM to allow more effective evaluation and communication of the maturity reached by a technology. In the following years, each TRL level was properly defined and since then, many organizations adopted this scale of values, and TRL proved to be highly effective in communicating the status of under-development technologies (Table 1).

Unfortunately, as it was highlighted in several references (Dowling et al, 2005), (Mankins, 2002), (Mankins, 2009), (Meyestrel, 2003), (Sauser, 2006), (Smith, 2005), (Valerdi, 2004), the TRL does not provide a sufficiently complete representation of the difficulty of integration of the under investigation technology within the system and it does not include any guidelines to cope with uncertainties that may be expected in moving through the TRL incremental path (Cundiff 2003), (Dowling et al., 2005), (Mankins, 2002),(Moorehouse, 2001), (Shishkio, et al. 2003), (Smith, 2005). Thus, when the TRL is abstracted from the technology level to a system or subsystem context, which may involve the cooperation of several different technologies, other indexes should be introduced. In particular, in 2006, (Mankins, 2009) in order to manage the increasing complexity of the aerospace products, the concept of System Readiness Level has been introduced with the aim of incorporating the already defined TRL scale with a new index, the Integration Readiness Level (IRL) (Kujawski, 2013). Following the definition proposed by Mankins, the IRL could be defined as a systematic measurement of the interfacing of compatible interactions for various technologies and the consistent comparison of the maturity between integration points (TRLs). In particular, Mankins suggested to exploit the IRL to describe the integration maturity of a developing technology with another technology, mature or already under-development. In particular, the following Table summarizes the definition envisaged for each IRL level.

Starting from these two indexes (i.e. TRL and IRL), the SRL can be derived, as a mathematical function of the TRLs in a system and their subsequent integration points with other technologies (IRL). From theses definition, 5 resulting SRL levels have been obtained and they are summarized, with their related definitions, in the following table.

Table 6: Technology Readiness Level (TRL) Definition

TRL	Definition
9	Actual system proven through successful mission operations
8	Actual system completed and qualified through test and demonstration
7	System prototype demonstration in relevant environment
6	System/subsystem model or prototype demonstration in relevant environment
5	Component and/or breadboard validation in relevant environment
4	Component and/or breadboard validation in laboratory environment
3	Analytical and Experimental critical function and/or characteristic proof-of-concept
2	Technology concept and/or application formulated
1	Basic principles observed and reported

Despite these three indexes are widely used among the scientific community, in the last few years, SRL and IRL have been criticized, mainly from the mathematical standpoint and some corrections have been suggested (Kujawski, 2013). However, all these indexes will support the designer, especially at the very beginning of the design process, and particularly in those strategic activities aimed at scheduling and planning activities.

Table 7:Integration Readiness Level (IRL) Definition

IRL	Definition
7	The integration of technologies has been verified and validated with sufficient detail to be actionable
6	The integrating technologies can accept, translate and structure information for its intended application.
5	There is sufficient control between technologies necessary to establish, manage and terminate the integration
4	There is sufficient detail in the quality and assurance of the integration between technologies
3	There is compatibility (i.e. a common language) between technologies to orderly and efficiently integrate and interact
2	There is some level of specificity to characterize the interaction (i.e. the ability to influence) between technologies through their interface
1	An interface (i.e. physical connection) between technologies has been identified with sufficient detail to allow characterization of the relationship

Table 8: System Readiness Level (SRL) Definition

SRL	Name	Definition
5	Operations and support	Execute a support program that meets operational support performance requirements and sustains the system in the most cost-effective manor over its total life cycle

4	Production and Development	Achieve operational capability that satisfies mission needs
3	System Development & Demonstration	Develop a system or increment of capability; reduce integration manufacturing risk; ensure operational supportability; reduce logistic footprint; implement human systems integration; design for producibility; ensure affordability and protection of critical program information; demonstrate system integration, interoperability, safety and utility.
2	Technology development	Reduce technology risks and determine appropriate set of technologies to integrate into a full system
1	Concept Refinement	Refine initial concept. Develop system/technology development strategy

2.2 Systems Engineering: an innovative approach for Aircraft Design

This section aims at providing the rationale laying behind the selection of a system engineering approach to cope with aircraft design activities in an innovative way. At the beginning, it could be very useful to notice that design, besides being an exciting, challenging, satisfying and rewarding activity, it should be considered a more advanced version of problem-solving technique (Sadraey, 2012). Differently from the general procedure for solving a mathematical problem, Design is not straightforward, being a much more subjective endeavour where a single "correct" answer is rarely present. Mathematical and scientific problems are well-posed in a compact form, meaning that the solutions to each problem are unique and compact, and they have an identifiable closure. However, a real-world engineering design problem does not share these characteristics, and it is usually not well-posed, i.e. it has not a unique solution, and open-ended. Following the definition of Engineering design proposed by the Accreditation

Board of Engineering and Technology (ABET), "*Engineering design is the process of devising a system, component, or process to meet desired needs. It is a decision making process (often iterative), in which the basic sciences and mathematics and engineering sciences are applied to convert resources optimally to meet a stated objective. Among the fundamental elements of the design process are the establishment of objectives and criteria, synthesis, analysis, construction, testing, and evaluation*."

In particular, a general design activity consists of three major tasks, recursively and iteratively carried out (Figure 38).

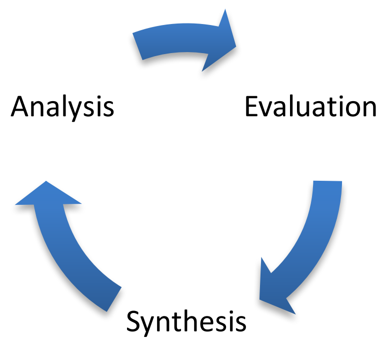


Figure 38: Design Activity Phases

- *Analysis*: is the process of predicting the performance or the behavior of a design candidate
- *Evaluation* is the process of predicting the performance calculation and comparing the predicted performance of each feasible design candidate to determine deficiencies.
- *Synthesis* refers to a combination of two or more entities that staying together form something new. This is the most challenging part of the design activity.

In order to cope with the high level of complexity, a System Engineering approach has been selected as basis for the development of a fully integrated design methodology. Indeed, starting from the identification of the major needs expressed by a set of stakeholders, a Systems Engineering approach allows to derive the system architectures starting from the describing the capabilities expected from the product or from its direct exploitation. This approach brings several benefits to the overall design process.

2.2.1 Traditional Systems Engineering approach: overview and application in aerospace design

In the last decade, the Systems Engineering (SE) approach has been widely used and also standardized. In particular ISO/IEC 15288 (ISO, 2002) is an international standard consisting in a generic process description. Notably, it identifies four process groups to support SE application during the overall systems life cycle (see Figure 39).

- ***Technical Processes*** include stakeholder requirements definition, requirements analysis, architectural design, implementation, integration, verification, transition, validation, operation, maintenance, and disposal.
- ***Project Processes*** include planning, assessment, control, decision-making, risk management, configuration management, and information management.
- ***Enterprise Processes*** include enterprise management, investment management, system life cycle processes management, resource management, and quality management
- ***Agreement Processes*** address acquisition and supply.

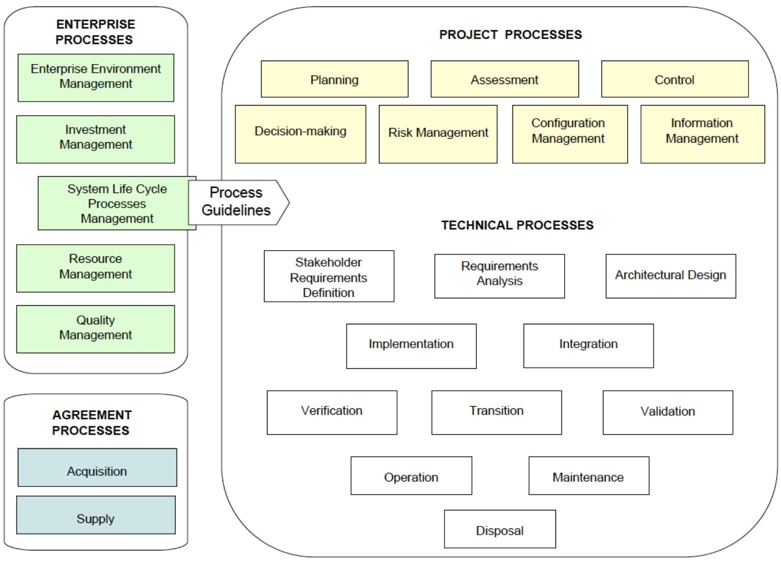


Figure 39: Systems Engineering Process Groups (ISO, 2002)

Besides several standardization attempts, a unique definition of Systems Engineering does not exist. However, the following three are the most representative of all the several aspects covered by this approach.

Systems engineering is a discipline that concentrates on the design and application of the whole (system) as distinct from the parts. It involves looking at a problem in its entirety, taking into account all the facets and all the variables and relating the social to the technical aspect. (Ramo, 2004)

Systems engineering is an iterative process of top-down synthesis, development, and operation of a real-world system that satisfies, in a near optimal manner, the full range of requirements for the system. (Eisner, 2008)

Systems engineering is an interdisciplinary approach and means to enable the realization of successful systems. (INCOSE, 2004)

Systems Engineering is based on Systems Thinking and this gives the practitioners a unique perspective on reality, training in recognizing the circular causation of the design activities, where a variable could be both the cause and the effect of another. Indeed, it is characterized by an iterative nature that supports learning and continuous improvement. Many benefits may arise from the application of a Systems Engineering approach, thanks to its ability of coping with complexity and changes that may affect the design activity.

One of the major benefit of the suggested approach is provided by the fact that the analyses start with a functional perspective before moving to the physical one, allowing the designers to derive a higher number of possible solutions for the problem. Besides the fact that a wider design space will imply major efforts (also from the computational standpoint) in dealing with it, this approach diminishes the risk of neglecting relevant and innovative solutions to be considered. Indeed, without a structured functional-based approach, it would be very difficult to have the possibility of considering, at least at mission level, all the possible candidate solutions, overcoming the problem of the development of new products only under the push of tradition.

Furthermore, the Systems Engineering approach provides a continuous support to enhance the traceability of each solution, creating clear relationships between stakeholders needs and design results. Of course, many other benefits are reported in the next sections.

The exploitation of SE approach can also reduce the risk associated with the development of innovative products. As far as risk is concerned, referring to (DAU, 1993), Figure 40 shows that the percentage of Committed Costs related to conceptual and preliminary design could overcome the 80% of the overall Life Cycle Cost (LCC).

Eventually, the application of a system engineering approach can significantly reduce the time from prototype to the market penetration of a new product.

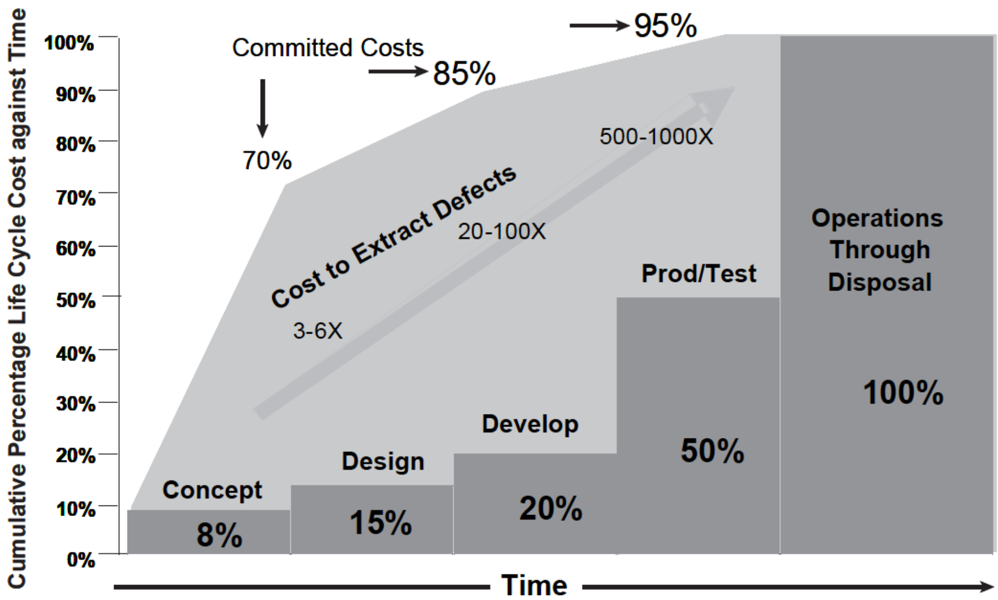


Figure 40: Committed Life Cycle Cost against time (DAU)

2.2.2 Towards a Model Based Systems Engineering Approach

Looking in particular to the need of the last decade and considering the very high level of integration required in the design of aerospace systems, and the relative increasing in complexity, the need of moving from a document-based approach to a model-based approach has been arisen (Fusaro, 2016).

Referring to (INCOSE, 2007) "*Model-based systems engineering (MBSE) is the formalized application of modelling to support system requirements, design,* *analysis, verification and validation activities beginning in the conceptual design phase and continuing throughout development and later life cycle phases*.”

In particular, accepting the definition of the INCOSE MBSE Focus Group (Estefan, 2007), Model-Based Engineering (MBE) is about elevating models in the engineering process to a central and governing role in the specification, design, integration, validation, and operation of a system. For many organizations, this is a paradigm shift from traditional document-based and acquisition lifecycle model approaches, many of which follow a “pure” waterfall model of system definition, system design, and design qualification. One of the biggest communication barriers that exists between the traditional engineering design disciplines (including the discipline of systems engineering) and MBE is that in a model-based process, activities that support the engineering process have to be accomplished through development of increasing detailed models (Hart, 2015).

The adoption of a MBSE approach carries several benefits to the design process:

- It allows anticipating specific discipline analyses during the conceptual and preliminary design phases, thanks to the exploitation of models that could be easily available since those high-level phases.
- It facilitates the introduction of changes, avoiding additional time and costs and providing a good way to trace changes allowing versioning and configuration management (also referred to Change and Configuration Management CCM).
- It allows sharing models, information and data among different design levels (*vertical sharing*) and among different involved disciplines (*horizontal sharing*).
- It facilitates requirements traceability, that could be expressed among requirements, belonging to different categories or hierarchical levels (*internal traceability*), but can also be highlighted between requirements and additional design items, that could be functions, variables, products, performances, etc. (*external traceability*) (Fusaro, 2016)
- Eventually, it allows the standardization and formalization of processes thanks to the ontology laying behind the models.

2.3 Integrated Design methodology for very complex and innovative aerospace products

The aim of this section is to provide an overview of the integrated methodology that has been conceived to carry out the aerospace system design and it is based on a MBSE approach. As secondary objective, this section should provide the readers with a basic understanding of the main differences of traditional and state of the art design methodologies with respect to the one developed all along this Thesis. Figure 41 provides a schematic view of the process that is currently implied in aircraft design (Raymer, 2012). Indeed, besides the fact that Multidisciplinary Design Optimization techniques are widely used in this field, improvements to the higher design phases, preparatory for the optimization cycles, are urgently needed to shorten the research and development time as well as to reduce the risk related to the introduction of new concepts.

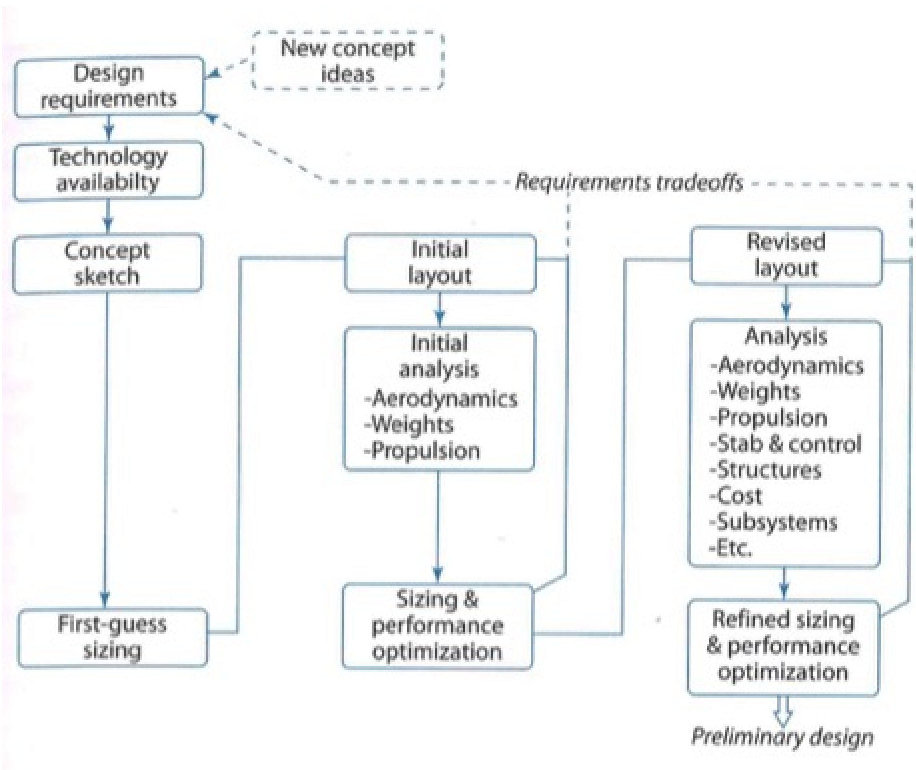


Figure 41: Overview of the integrated design methodology

Analyzing Figure 41, it can be noticed that the generation of new concepts is mainly imposed from the outside, so that the overall process aims at developing this new concept without having a clear idea of the reason why. The drivers that led to the generation of the concepts are not considered since they are provided from previous assessments without any specific traceability process. For this reason, crucial points like the need for innovation, or, simply, its basic characteristics do not enter within the design cycle but are embedded within the concept itself. Besides this process undoubtedly presents important advancements with respect to the oldest design procedures, where qualitative and quantitative evaluations were carried out in a sequential order, Figure 41 reveals that the entire workflow is strongly influenced by the technology availability. This produces, as a result, a product developed in a technology-pushed environment, while the mission is only considered in a later stage. Moreover, the initial layout definition mainly relies on suggestions coming from previous studies that could not consider the need of integrating innovative technologies on-board and the satisfaction of the different stakeholders' requirements is verified at the end of each design stage, enhancing the number of recursions and iterations. The present work tries to overcome these highlighted criticalities, suggesting an innovative conceptual design methodology able to be applied to the design of very innovative aerospace products.

2.3.1 Integrated Methodology overview

Figure 42 provides a general overview of the innovative methodology that has been developed to improve the current approaches (see Figure 41). Besides the development time, reported horizontally, it could be worth noticing that the methodology deals with different design layers. The picture describes the conceptual and preliminary phases only, but one of the major benefits is that the proposed approach could be extended to the overall product life-cycle. As it is clearly explained later on in this, and even more in the next chapters, the high level of integration requires a multidisciplinary approach that imposes to anticipate analyses at system and subsystem levels since the conceptual and preliminary design phases. Differently from what suggested in Figure 41, the general perspective is completely different. In Figure 42, the overall process is mission-pulled, and technologies are in-depth analyzed, considering their impact on the layout since the very beginning, even if they are dealt with in their proper hierarchical level. Moreover, the innovative concepts are not superimposed but a wide range of possible innovative alternatives are derived elaborating the ideas and the desires expresses by stakeholders, taking into account at the same time suggestions coming from the market outlook and restrictions from regulations.

The impact of stakeholders and all the other high level inputs are traced all along the project and they are never considered embedded but clearly highlighted, especially when used in selection criteria. A specific tool-chain able to support the overall process is envisaged and it will allow a complete traceability. The tool-chain has been developed taking also into account the current need of increasing in automation especially envisaged by key-players in the aerospace industrial domain. Thus, the traceability is not only considered a major benefit in terms of design but also from a management standpoint. Eventually, the methodology presented in this Thesis has the ambitious goal of providing a unique structured set of guidelines to support the vehicle layout definition. Indeed, in a connected workflow, the different views of the sketch will be drafted step-by-step following a rationalized procedure in which the value of each design parameter will be the expression of a higher stakeholders' need or regulation suggestion. This high level of traceability facilitates the generation of parametric CAD models and different simulation models easily to keep updated and paves the way to the future introduction of Multidisciplinary Design Optimization tools in the workflow.

In the following paragraphs, a general description of the methodology is reported.

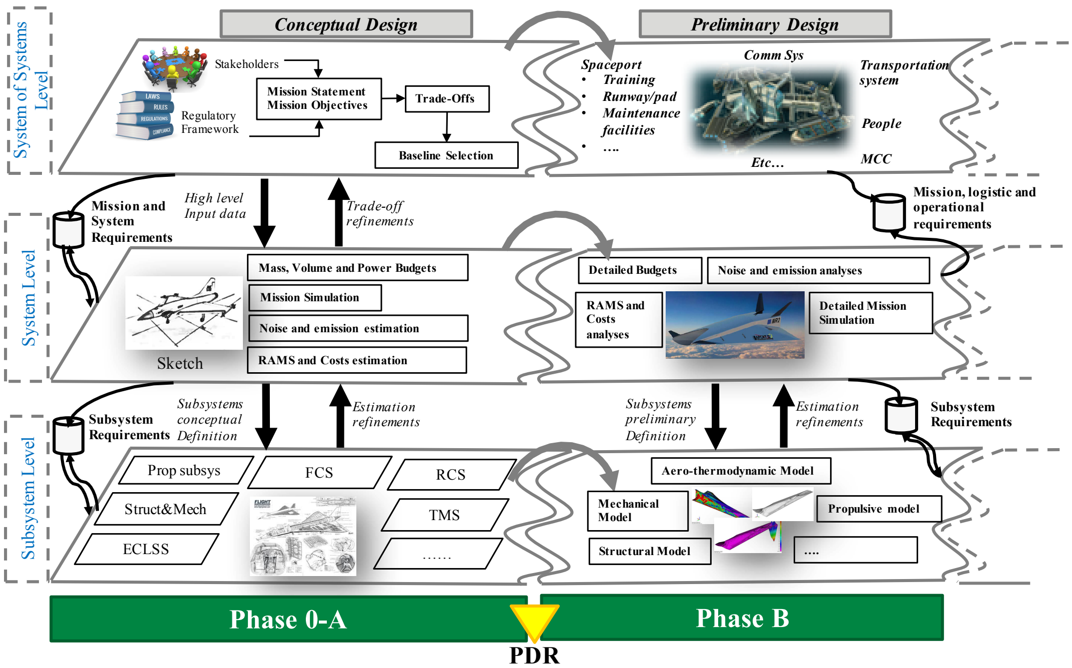


Figure 42: Overview of the integrated design methodology

2.3.1.1 From the stakeholder analysis to the mission statement

The design process should start with the identification of the stakeholders and their needs because it is the starting point to define the major objectives of a project. The stakeholder analysis together with a deep insight into the market forecast provide the basic elements to build up a proper business case. Moreover, for completeness, it is worth considering the regulatory framework in which the project should be carried out and in which the transportation system will be operated. Indeed, this activity would derive a first draft list of operational requirements and constraints, contributing to arise some relevant safety issues to be taken into account since the beginning of the design process. Theoretically, stakeholder's analysis, market forecast and regulatory framework analysis are the basic analyses that should be performed in order to derive a well-defined project statement. However, in a real world, the objectives of each initiative are deeply affected by some high-level strategic considerations. Indeed, especially for projects involving stakeholders like governments and space agencies, economic and political constraints should be considered. At this purpose, as it will be in-depth described in detail in the next chapter, in order to couple technical and technological issues with strategical considerations, a technological roadmap should be derived. It is then possible to define the mission statement of the different incremental steps required for increasing the readiness level of the enabling technologies, considering the existing budget and political constraints.

As it has been stated since the beginning of this chapter, in order to cope with need of enhancing the overall design process, a support tool-chain has been envisaged. In particular, to support the stakeholder analysis, a proper automatized version of the Quality Functional Deployment Tool (QFD), also known as House of Quality, has been built. This is a very useful tool allowing to translate the customers' needs into useful design parameters. It consists of a series of scoring assessments through which it is possible to set up a prioritized lists of stakeholders' needs and to formalize the major technical problems that would be especially considered. In this case, a proper tool has been developed using Matlab® and MS Excel® platforms. Conversely, the strategic decisions could be properly supported by a Technology Roadmap Generation Tool directly linked to a Database collecting all the past and currently under-development initiatives in a specific field. The crucial role of Database in such integrated design processes with respect to its relationship with the proposal of a roadmap would be stressed later on. Furthermore, the role of the Database will also be investigated in different phases of the design process, especially supporting statistical analyses. Both for the roadmap tool and for the supporting database, Matlab® and MS Excel® have been exploited.

2.3.1.2 From the mission statement to the mission baseline selection

Once the mission statement is derived, major objectives can be easily listed and a first list of requirements (usually mission and programmatic) could be derived. From this point it is possible to start with the mission analysis. This high-level analysis aims at freezing the mission baseline considering the overall System of Systems. In particular, once the main objectives of the mission have been clarified, the developers should elaborate different ideas to accomplish the enviable mission. Exploiting the benefits coming from the enhancement in autonomous computing, in the last decade, the possibility of postponing the trade-off later on in the projects arose, allowing the designers to manage with a higher number of solution alternatives. In particular, to have the possibility of considering all the feasible alternatives, a system engineering approach could be successfully applied, since this high design level. For the purpose of this work, and considering the available literature, this set of activities could be roughly divided into a Functional Analysis part and a Concept of Operations (ConOps) part. This approach gives a rationale to the very first brainstorming activities in supporting the generation of mission concept alternatives. From one side, the functional analysis allows looking at the objectives to be achieved from a pure functional standpoint and to gradually moving to the physical perspective with the aim of defining the major components able to perform the functions, deriving all the possible alternatives to carry them out. From the other side, ConOps allows looking at the SoS under definition from a behavioral and operational standpoint. To carry out these two steps, typical tools developed and used for the functional analysis can be exploited. For example, the functional tree can allow defining the main functions the mission shall perform and a function/product matrix could help to structurally define the variety of elements able to accomplish the previously deduced functions (Viscio, 2015), (Fusaro, 2017).

At this stage, a functional tree expresses the functions to be performed for the execution of the mission. The functional tree allows splitting the higher level and complex functions, which stem from the mission objectives, into lower level functions, through a typical breakdown process, eventually allowing the identification of the basic functions that have to be performed by the future product. Therefore, starting from the so-called top-level functions, the functional tree generates various branches, moving from the most complex functions to the basic functions, i.e., those functions at the bottom of the tree that cannot be split any further. The basic functions help defining the functional requirements of the future product, as each basic function can be rewritten as a functional requirement.

A function/device matrix allows identifying the elements or building blocks needed to accomplish the functions. Specifically, matrix's rows contain the basic functions coming from the functional tree, while columns report the products, i.e., the space mission elements capable of performing those functions. Starting from the analysis of the first basic functions, new elements progressively fill in the columns. Eventually, all basic products are determined. As a result, the elements to be involved in the missions are identified, by mapping all basic functions to products.

One of the major activities is to group and combine the elements to derive the different mission concept options. During this process, it is also important to evaluate how well each of the different options of each single function is able to accomplish the function itself and which is its relationships with the other functions of the mission. To increase the level of autonomy of the process, the authors suggest to use the quality function deployment (QFD) tool, also known as house-of-quality. In particular, in this context a modified version of the basic QFD (presented and exploited in support to the stakeholder analysis) has been created and applied to a specific case study

Product tree can be obtained grouping together the elements identified in the function/product matrix. Unlike the functional tree, which has a typical top-down approach, the development of the product tree follows a straightforward bottom-up process. Like in the functional tree, also in this case, it is extremely important to clearly define the level of decomposition at which each product belongs to.

Block diagrams represent the building blocks linked through point-to-point connections. The block diagram provides the designer with further information, if compared to the connection matrix, about the links' directionality. Moreover, it gives evidence of the type of connections (e.g., mechanical, electrical, etc.). From these diagrams, configurational requirements can be refined and interface requirements can be derived.

Considering the ConOps, Functional flow block diagrams (FFBD) allow defining the different operations the system shall perform and the different phases and operative modes. FFBDs specifically depict each functional event (represented by a block) occurring, following the preceding function. Some functions may be performed in parallel, or alternative paths may be taken. The FFBD network shows the logical sequence of "what" must happen; it does not ascribe time duration to functions or between functions. The FFBDs are function oriented, not equipment oriented.

Complementary, to derive possible mission concepts, it is also important to describe the system from an operative point of view. At this first level, the concept of operations consists in hypothesizing the general way of working of the systems, including evaluations of mission phases, operation timelines, operational scenarios, end-to-end communications strategy, command and data architecture, operational facilities, integrated logistic support, and critical events. In fact, according to NASA Handbook (NASA, 2007), the ConOps is an important component in capturing stakeholder expectations, requirements, and the architecture of the project. It stimulates the development of requirements and architecture related to the user elements of the system. It serves as the basis for subsequent definition documents, such as the operations plan, launch and early orbit plan, and operations handbook, and provides the foundation for the long-range operational planning activities, such as operational facilities, staffing, and network scheduling.

It is clear that an extremely high level of uncertainty characterizes preliminary design stages. To mitigate this problem, an ad-hoc tool has been built to simulate the mission, starting from a limited set of inputs. The results, although characterized by a high degree of approximation, can allow estimating the very first quantitative data, and they can be exploited to evaluate figures of merit in the trade-off analysis. Please notice that one of the major objective of the present work is connecting together, all these tools above presented, integrating them within a structured tool-chain following a MBSE approach. In particular, exploiting the functionalities at the beginning of this process, a Use Case Diagram (UCD) can be exploited to support and formalize the mission statement and the related mission objectives, taking into account all the sources of needs, desires or constraint to the mission definitions. Indeed, considering that for definition, each use case is defined as the objective that each actor wants to reach, in the aerospace context there is a perfect analogy with mission objectives and mission statement (seen as a formalized collection of mission objectives and constraints). It is also clear that it is necessary to deal with requirements, since this high-level design phase, envisaging a proper organized and structured requirements database. For instance, the authors selected IBM Rational Doors ® to store and manage Requirements and IBM Rational Rhapsody® to carry out the other analyses of this phase. Since the beginning, it is necessary to guarantee a complete traceability between requirements and between requirements and other elements of the models, as for example mission objectives. In the rest of the paper, the authors will refer to the first type of traceability as *internal traceability*, while in all the other cases in which the requirement is related to another model's element, the expression *external traceability* will be exploited. Moreover, as it will be discussed, it is also very important to guarantee complete traceability among requirements belonging to different levels or types. In this case, the term internal traceability will be used. Once the main mission objectives have been derived, tools are required to support the preliminary definition of the system and its related subsystems and components from the functional, physical and operative standpoint. Starting from the functional standpoint, after the identification of the top-level function, directly related to the mission objectives, it is possible to generate functional trees exploiting Block Definitions Diagrams (BDDs) whose functions represent the first draft list of functional requirements. Considering these requirements from a formal point of view, they are statements whose subject can only be a generic system. Then, moving to a physical standpoint, the creation of function/devices matrixes can enable to precisely identify the subject of these statements and, in this way, the first list of functional requirements can be updated. In addition, BDDs can be exploited again to build product trees and the generated elements can be considered to point out the mutual relationships among each other, using Internal Block Diagrams (IBDs). In order to enhance traceability, each identified product is directly connected to the related functional requirement. This link allows to understand whether or not a product satisfies a certain requirement (external traceability). Moreover, the structure of the IBD can allow to generate a list of interface requirements. At this stage, the designer should consider the system from an operational perspective starting with the definition of Functional Flow Block Diagrams (FFBD) exploiting SysML Activity Diagrams (ADs) and based on the functions contained in the BDD for the functional tree. Then, Mission Phases and Modes of Operations should be identified and connected to the other elements of this integrated system. In this case, UCD, Sequence Diagrams (SDs), and State Machines Diagrams (SMDs) can be exploited. In particular, the Use Case Diagram can have a slightly different meaning depending on the phases design level. Indeed, at system level, each use case can embody a phase of the product life cycle and at subsystem levels and at lower levels, the use cases can be representative of a mission phase. It is worth to notice that each phase can be connected to all those functions of the tree that are required for that specific part of the mission. This can be formalized using dependency links (Figure 43).

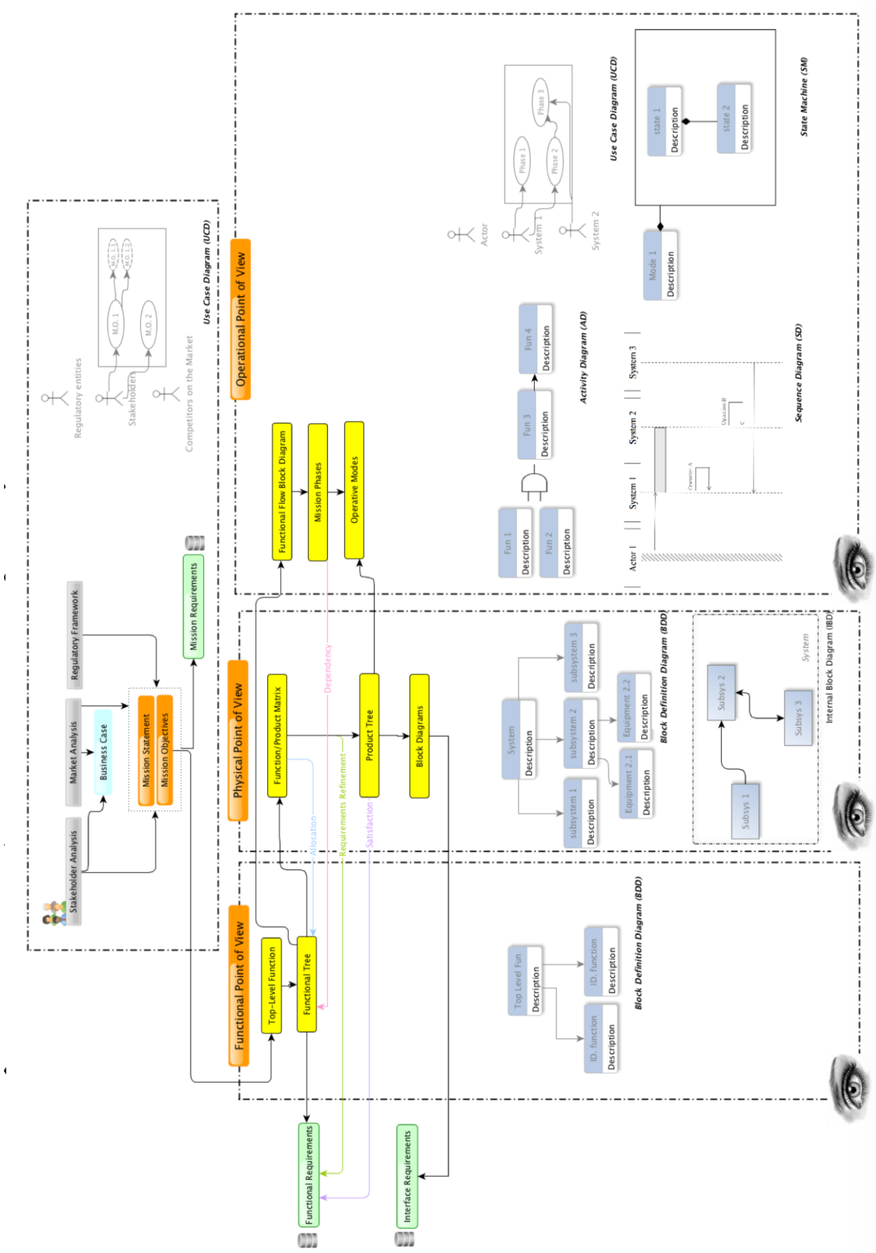


Figure 43: System Level Analysis SE toolchain (ISO, 2002)

During the overall design process, important moments are those in which decisions should be taken, as for example, the selection of the most promising alternative among the list of possible candidates elicited thanks to the exploitation of a SE approach.

In order to move from a pure qualitative approach to a more rationalized one, in which each design choice has a reasonable background of motivations in support, a precise activity of decision making inspired to the Analytical Hierarchy Process (AHP) has been formalized (Figure 44). In addition, when it has been possible to link the judging criteria to some numerical values, a series of parametric equations to support the trade-off have been developed.

While the Trade-off analysis technique is based on the evaluation of some candidate alternatives on the basis of measurable parameters, different decision making processes have been investigated in order to allow formalizing judgments. At the end, AHP is an organized way to make decision generating a list of priorities through pairwise comparisons. Referring to Saaty (Saaty, 2008), who formalized this approach, the AHP consists of four different steps: problem definition, definition of the decision hierarchy structure, comparison matrices evaluations and prioritization of the alternatives. As it will be clearly shown in some applicative examples, the crucial aspect will be the definition of a proper fundamental scale of scores.

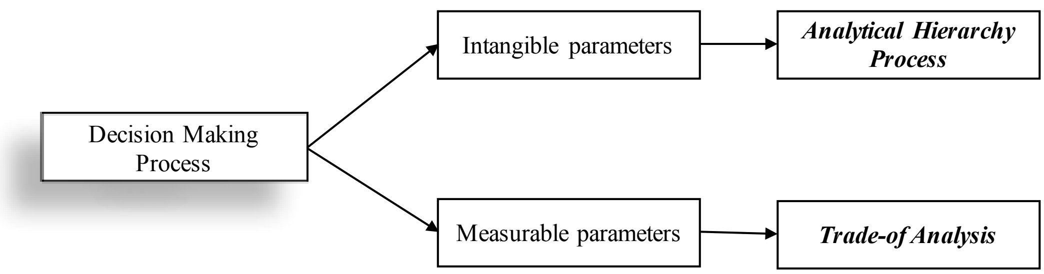


Figure 44: Decision Making Process

2.3.1.3 From mission baseline selection to the transportation system design

Once the mission has been set up, it is possible to move to the systems design, having a consolidated list of requirements, constraints and a "model" of the mission too. For completeness, moving from mission level to system level, all the mission segments should be considered. Besides the fact that from here onward the design activity focuses mainly on the transportation system and related subsystems, it is important to notice that the same methodology may be applied to the conceptual and preliminary design definition of the other mission segments, such as for example the spaceport (Santoro, 2013) (Santoro, 2016). This is a very important capability of the tool-chain allowing the possibility of tracing the impact of the other elements of the mission on the transportation system and vice-versa.

The diagram in Figure 46 focuses on the transportation system and shows that the first activity to be carried out should be the architectural layout definition. In this context, the engineers should start with the selection of the most suitable staging and propulsive strategies and, from these analyses, feasible take-off and landing strategies should be hypothesized. Then, to complete the selection of the optimal aerothermodynamic layout for each stage should be performed. This set of activities has always been very qualitative besides their absolute relevance in the overall design and development process but the alternative selection can be carried out in formal way through the application of decision making process.

Once the architecture has been defined, the design of the transportation system should be started. Following a process similar to the one exploited to derive the mission baseline, a Systems Engineering approach has been used to support the design at system level. The exploitation of functional analysis will allow describing the system through to the elicitation of a new set of requirements and the identification of the major systems to be investigated in the next iterations. In addition, the exploitation of the ConOps will allow considering high level connections among the different systems and starting the definition of possible modes of operations. In particular, it has to be noticed that an important role will be played by the simulations. At system level, to carry out mission simulations, ad-hoc built-in tools or commercial software (like ASTOS® or STK®) can be exploited during the very first design iterations. In the next iterations, when more detailed functional and physical models will be available with a sufficient levels of detail, additional simulation tools could be exploited such as Simulink® or

CAD/CAE software. In particular, in the following chapters, the role of each simulation tool within the aircraft design workflow will be clearly outlined, with special emphasis on the way in which the outputs could automatically feed a requirements elicitation or updating process. In this context, it will also be clearer the exploitation of a MBSE approach.

Then, the designer should concentrate on the design stage and then moving to the subsystems and components design. In particular, as far as it is shown in the scheme of Figure 46, the design activity follows the same basic activities carried out at the higher levels. At the beginning, it is necessary to evaluate the most suitable alternative architecture and perform the first high-level estimations. This activity is a fundamental step in aircraft design and it could be supported by the use of statistics. However, in case of hypersonic initiatives or, in general, of innovative configurations, statistical data could be very difficult to be collected. In particular, in the case of hypersonic transportation system, in the past, specific formulations allowing mass and volumes estimations at this high level of design have been suggested. HASA (Harloff, 1988), for example provides useful suggestions but due to the years in which it has been developed, this conceptual design methodology is based on an aged statistical basis. At this purpose, during this research, great attention has been devoted to the creation of a Database for Hypersonic initiatives (HyDat) (Fusaro, 2017b). The presence of a database guarantees the possibility of having continuously updated coefficients to be used within the statistics-based estimations.

Moreover, mass estimations are absolutely crucial at this level of design but, it is also important to have a first estimation of the other design characteristics, such as Lift-to-Drag ratio, wing surface, aerodynamic coefficients, etc. Also at this purpose, HyDat could be usefully applied. In particular, the author exploits it to provide a list of preliminary estimation equations, in which the values of the parameters should have been properly selected, adequately filtering the existing data. Then, depending on the considered subsystem, ad-hoc sizing algorithms specifically envisaged for conceptual and preliminary design phases have been developed. In particular, as it is clearly detailed in each Chapter devoted to sizing, the different workflows are characterized by the capability of starting from high level estimations in order to derive more precise evaluations. All these processes are carried out in a MBSE environment in which functional, physical and behavioral models are continuously updated as well as the lists of requirements. In addition, special attention will be devoted to guarantee the traceability of all the design choices with respect to the top-level stakeholder needs.

Furthermore, the results obtained at system level will be used to carry out high level estimations in terms of Reliability, Availability, Maintainability and Safety (RAMS) and costs. These activities have always been considered as secondary and they were carried out at the end of the design process, when the configuration is fixed and sufficiently detailed. In this methodology, they are considered of primary importance because, especially for highly innovative configurations, they can generate a set of important requirements and stringent constraints or they can show the need of heavy changes in the aircraft layout. Remembering the high percentage of committed costs defined during the conceptual design phase, reported in Fig. #, it is convenient to carry out these first RAMS and costs estimations as soon as possible in the design cycle.

Then, it is possible to continue with the analyses at subsystem and component levels until reaching the technology level. At the end of this iterative process, once the Technology level is reached, a process of technology roadmap refinement could be started with the possibility of in-depth envisaging the future steps required for the development and verification process, taking into account the different readiness levels of each identified enabling technology.

The overall process that has been sketch in this chapter and summarized in the following two Figures, will be in-depth analyzed in the next chapters.

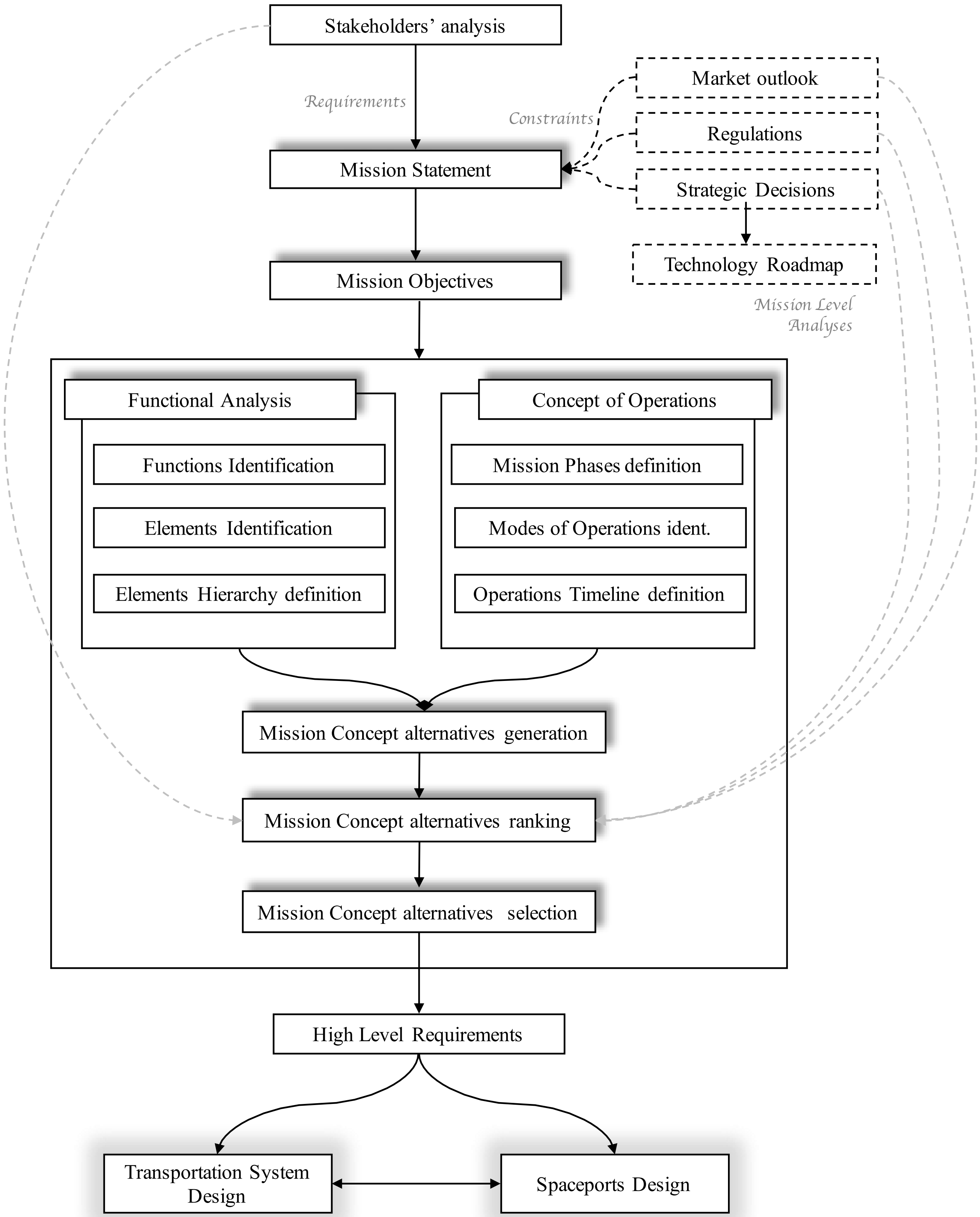


Figure 45: Methodology flow-chart: from stakeholder analysis to the Mission Baseline Selection

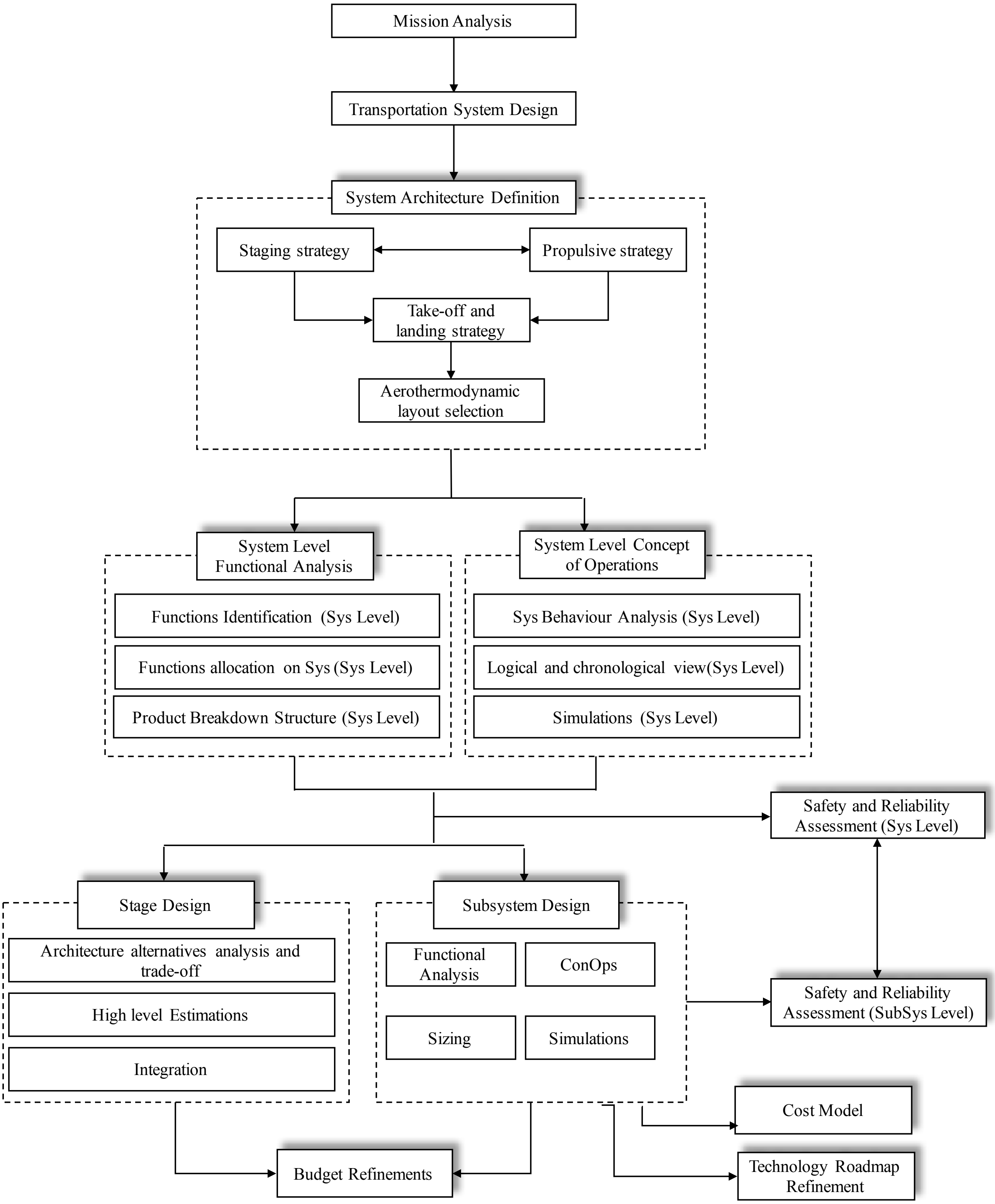


Figure 46: Methodology flow-chart: from Mission Baseline Selection to technology level analysis

References

Cundiff, D. (2003). Manufacturing Readiness Levels (Mrl).Unpublished white paper.

Defense Acquisition University, (1993), Committed Life Cycle Cost against Time.

Dowling, T., and T. Pardoe (2005). Timpa - Technology Insertion Metrics Volume 1." edited by Ministry of Defence: QinetiQ, 2005.

Eisner, Howard (2008). Essentials of project and systems engineering management. *John Wiley & Sons*.

Estefan, Jeff A. (2007). Survey of model-based systems engineering (MBSE) methodologies. Incose MBSE Focus Group 25.8 (2007).

Fusaro, R., Ferretto D., and Viola N. (2016). Model-Based Object-Oriented systems engineering methodology for the conceptual design of a hypersonic transportation system. *Systems Engineering (ISSE), 2016 IEEE International Symposium on. IEEE*, Edimburgh (Great Britain).

Fusaro, R., Viola, N., Fenoglio, F., Santoro, F. (2017). Conceptual design of a crewed reusable space transportation system aimed at parabolic flights: stakeholder analysis, mission concept selection, and spacecraft architecture definition. *CEAS Space Journal*, 9 (1), pp. 5-34. DOI: 10.1007/s12567-016-0131-7

Fusaro, R., Aleina, S.C., Viola, N., Longo, J., Saccoccia, G. (2017b) Database on hypersonic transportation systems: A versatile support for the technology roadmap generation and conceptual design activities. *21st AIAA International Space Planes and Hypersonics Technologies Conference, Hypersonics* 2017, 16 p.

Harloff, Gary J., and Brian M. Berkowitz (1988). HASA: Hypersonic Aerospace Sizing Analysis for the Preliminary Design of Aerospace Vehicles.

Hart, Laura E. (2015) Introduction To Model-Based System Engineering (MBSE) and SysML Presented at the *Delaware Valley INCOSE Chapter Meeting* July 30, 2015

INCOSE (2004), Systems Engineering Handbook, version 2a, June 2004.

INCOSE (2007) INCOSE SE Vision 2020 (INCOSE-TP-2004-004-02, Sep 2007)

ISO/IEC 15288: 2002(E), Systems engineering – System life cycle processes, Geneva: *International Organization for Standardization*, issued 1 November 2002

Kujawski, Edouard. (2013). Analysis and critique of the system readiness level. *IEEE Transactions on Systems, Man, and Cybernetics*: Systems 43.4: 979-987.

Mankins, J.C. (2002) Approaches to Strategic Reseach and Technology (R&T) Analysis and Road Mapping." *Acta Astronautica* 51, no. 1-9: 3-21.

Mankins, J. C. (2009). Technology readiness assessments: A retrospective. *Acta Astronautica*, 65(9), 1216-1223

Meystel, A., J. Albus, E. Messina, and D. Leedom. (2003). Performance Measures for Intelligent Systems: Measures of Technology Readiness. *PERMIS '03 White Paper*.

Moorehouse, D.J. (2001) Detailed Definitions and Guidance for Application of Technology Readiness Levels." *Journal of Aircraft* 39, no. 1 (2001): 190-92.

NASA, (2007). NASA Systems Engineering Handbook." National Aeronautics and Space Administration, *NASA/SP-2007-6105 Rev1*.

Ramo, Simon (2004). Systems Engineering Manual. *Federal Aviation Agency* (FAA).

Saaty, Thomas L. (2008) Decision making with the analytic hierarchy process. *International journal of services sciences* 1.1 : 83-98.

Sadraey, Mohammad H. (2012). Aircraft design: A systems engineering approach. *John Wiley & Sons*,.

Santoro, F., Del Bianco, A., Viola, N., Fusaro, R., Albino, V., Binetti, M., Moretti, F. (2016) Advances in Italian spaceports identification as infrastructures for suborbital flight activities and mission profiles.

Proceedings of the International Astronautical Congress, IAC, Guadalajara (Mexico).

Sauser, B., Verma, D., Ramirez-Marquez, J., & Gove, R. (2006). From TRL to SRL: The concept of systems readiness levels. *In Conference on Systems Engineering Research*, Los Angeles, CA.

Shishkio, R., D.H. Ebbeler, and G. Fox. (2003). Nasa Technology Assessment Using Real Options Valuation. *Systems Engineering 7*, no. 1 (2003): 1-12.

Smith, J.D. (2005). An Alternative to Technology Readiness Levels for Non-Developmental Item (Ndi) Software. Paper presented at the *38th Hawaii International Conference on System Sciences*, Hawaii.

Valerdi, R., and R.J. Kohl. (2004). An Approach to Technology Risk Management. Paper presented at the *Engineering Systems Division Symposium*, Cambridge, MA, March 29-31.

Viscio, M.A., Viola, N., Fusaro, R., Basso, V. (2015) Methodology for requirements definition of complex space missions and systems. *Acta Astronautica*, 114, pp. 79-92. DOI: 10.1016/j.actaastro.2015.04.018

Chapter 3

From Stakeholders Analysis to the Mission Baseline Selection

3.1 Mission Analysis process overview

This chapter focuses on the very first part of the integrated design methodology that aims at deriving Mission Concepts and Architectures starting from very high level analyses. Starting from an in-depth study of the Stakeholders aimed at understanding who they are and what are their major needs, the hypothetical scenario is bounded by considering the trends of the latest market forecasts and the regulatory framework in which the project is supposed to be operated. In addition, in order to define and formalize the purpose of the mission through the Mission Statement, possible constraints coming from high level strategic decisions should be considered. Then, once the major objectives have been derived, following a Systems Engineering approach, functional analysis and Concept of Operation techniques can be exploited to generate a very high number of possible ways to accomplish the envisaged objectives. Of course, the feasibility of each concept should be properly investigated with a consequent pruning of all the generated alternatives. It is clear that the methodology leads the designers to move from a qualitative to a more quantitative approach, as soon as the first data become available. For this reason, trade-off analyses with the support of simulations is proposed and integrated within the workflow summarized in Figure 47. This flow-chart, already presented at the end of the previous Chapter, is here reported as easy support that aims at summarizing the major steps of the process suggested in this chapter, providing also useful elements to understand the major relationships of the activities analysed in this contest with those one that will be carried out following design steps.

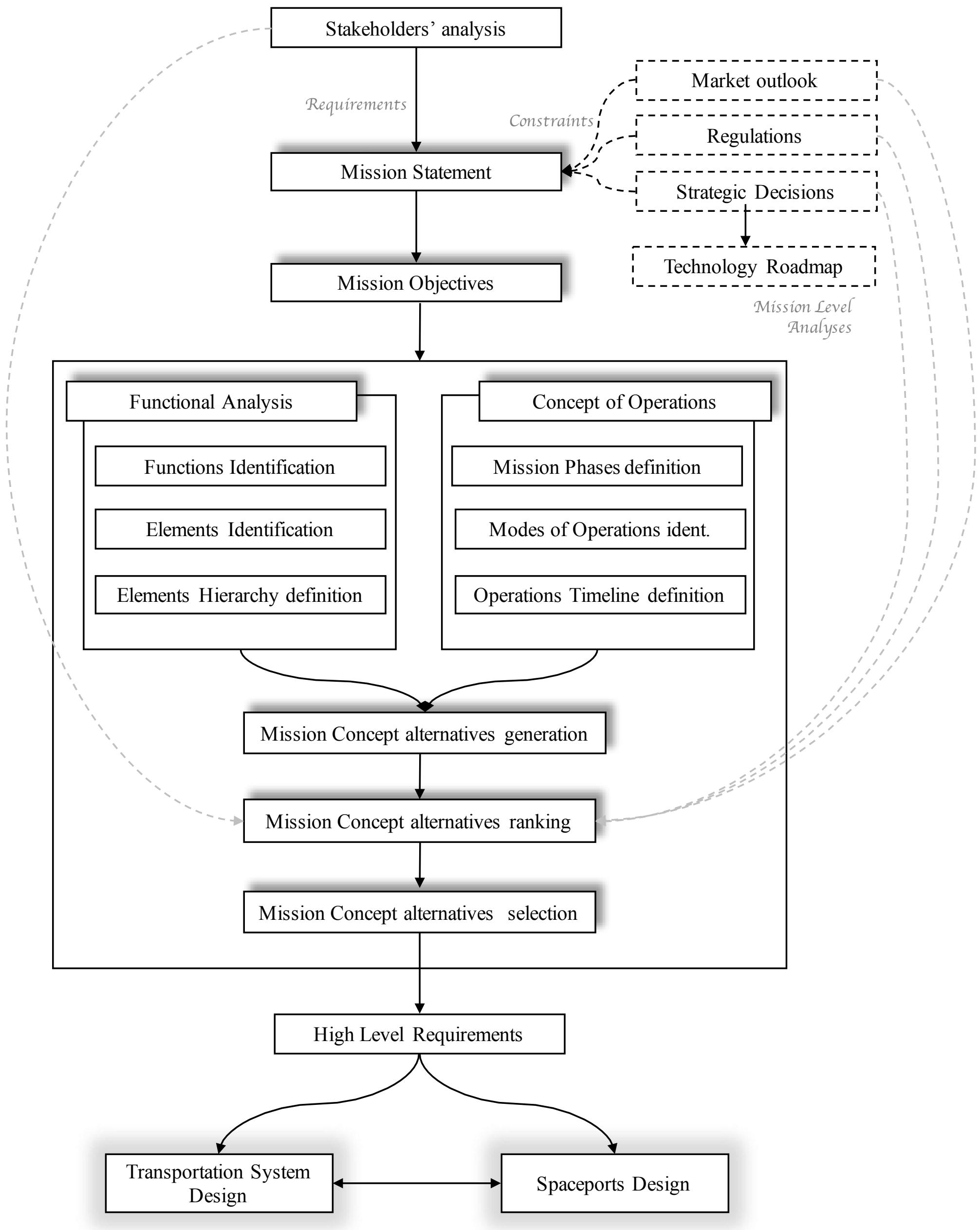


Figure 47: Methodology flow-chart: high level analyses

3.2 Stakeholders’ analysis and market forecast

3.2.1 Stakeholders and needs identification for hypersonic initiatives

The overall design process is intended to start with the identification of the Stakeholders, i.e. all those people that could be interested in the project or in the service. In order to pursue a well-organized stakeholder analysis, at first, it is important to understand the role that each identified stakeholder could play in the specific mission. Then, it is necessary to gather their needs and discover their hidden desires. In order to obtain a list of categorized needs and expectations for the project, a process based on the exploitation of Quality Functional Deployment Tools (QFD) in-cascade have been envisaged.

Following the NASA guidelines for classification (NASA, 2007), stakeholders could be classified depending on their role and interest in the project, as follows:

- *Sponsors*: private or public associations who establish a mission statement and fix boundaries on both the schedule and funds availability.
- *Operators*: all those people, usually belonging to engineering associations, in charge of controlling and maintaining both space and ground assets.
- *End-users*: all those people that will receive benefits from the mission operations and will use space mission’s products or capabilities. Usually they belong to the scientific or engineering community.
- *Customers*: differ from the previous category, because they are users who pays fees to exploit specific products or services offered by the mission.

Considering the specific case of the development of a hypersonic transportation system, different stakeholders could be identified for each of these categories. The following tables (Table 9,

Table **10**, Table 11, and Table 12) aim at providing an exhaustive overview of the major players in the context of the hypersonic transportation system. In addition, they suggest a possible list of interests, needs and desires that could be envisaged in the wide field of hypersonic, collecting both the cases of suborbital touristic flights and point-to-point connections. In order to have a wider and deeper understanding, the authors analysed not only the Italian and European contexts, but took also a look at the overall International framework. In particular, the author has been part of a research Italian boarding aims at providing an overview of the different initiatives, research or simply activities currently on-going in the Italian territory and guided by the Italian Centre for Military Studies. Moreover, these tables benefit from the strict cooperation between Politecnico di Torino and the European Space Agency (ESA) within a research activity framework aimed at developing a Database collecting worldwide Hypersonic Initiatives.

Table 9: Stakeholders’ needs in hypersonic initiatives: SPONSORS

Stakeholders’ category: SPONSORS	
Stakeholders’ identification	**Stakeholders’ expectations**
National and International Space Agencies (ESA, NASA, JAXA, CSA, etc…)	• To demonstrate technical superiority in the field of hypersonic speeds • To develop hypersonic entry and re-entry vehicles • To produce hypersonic entry and re-entry vehicles • To operate hypersonic entry and re-entry vehicles • To enhance the public consensus in space initiatives • To enhance reusable access to space • To foster international cooperation
Private enterprises	• To enhance reusable access to space • To develop transportation systems aimed at performing touristic parabolic flights. • To produce transportation systems aimed at performing touristic parabolic flights. • To commercialize transportation systems aimed at performing touristic parabolic flights. • To operate transportation systems aimed at performing touristic parabolic flights. • To develop transportation systems aimed at performing point-to-point hypersonic connections. • To produce transportation systems aimed at performing point-to-point hypersonic connections.

	• To commercialize transportation systems aimed at performing point-to-point hypersonic connections. • To operate transportation systems aimed at performing point-to-point hypersonic connections. • To develop, produce, commercialize and operate transportation systems aimed at guaranteeing reusable access to space. • To develop transportation systems aimed at guaranteeing reusable access to space. • To produce transportation systems aimed at guaranteeing reusable access to space. • To commercialize transportation systems aimed at guaranteeing reusable access to space. • To operate transportation systems aimed at guaranteeing reusable access to space.
Civilian aeronautical agencies	• To develop transportation systems aimed at performing point-to-point hypersonic connections. • To produce transportation systems aimed at performing point-to-point hypersonic connections. • To operate transportation systems aimed at performing point-to-point hypersonic connections.
Military aeronautical agencies	• To demonstrate national superiority • To develop transportation systems aimed at performing touristic parabolic flights. • To produce transportation systems aimed at performing touristic parabolic flights. • To operate transportation systems aimed at performing touristic parabolic flights. • To develop transportation systems aimed at performing point-to-point hypersonic connections. • To produce transportation systems aimed at performing point-to-point hypersonic connections. • To operate transportation systems aimed at performing point-to-point hypersonic connections.
Regulatory Entities (e.g., ICAO, ENAC, EASA, FAA,	• To develop a new regulatory framework for private commercial space activities. • To develop a new regulatory framework for parabolic flights.

etc.)	• To develop a new regulatory framework for hypersonic point-to-point flights.

Table 10: Stakeholders’ needs in hypersonic initiatives: OPERATORS

Stakeholders’ category: OPERATORS

Stakeholders’ identification	**Stakeholders’ expectations**
Private companies	• To provide routine hypersonic point-to-point flight connections. • To provide routine hypersonic parabolic flights. • To provide scheduled hypersonic space access. • To design, develop and manage spaceports and logistics facilities
Public companies	• To provide routine hypersonic point-to-point flight connections. • To provide routine hypersonic parabolic flights. • To provide scheduled hypersonic space access. • To design, develop and manage spaceports and logistics facilities
National and International Space Agencies (ESA, NASA, JAXA, CSA, etc…)	• To increase the possibility of space access.

Table 11: Stakeholders’ needs in hypersonic initiatives: END-USERS

Stakeholders’ category: END-USERS

Stakeholders' identification	Stakeholders' expectations
Private companies	• To receive, analyse and exploit data coming from the mission of from scientific experiments carried out on board
National and International Space Agencies (ESA, NASA, JAXA, CSA, etc…)	• To receive, analyse and exploit data coming from the mission of from scientific experiments carried out on board • To enhance the public consensus about commercial flight initiatives.
Scientific and engineering communities	• To receive, analyse and exploit data coming from the mission of from scientific experiments carried out on-board • To reach higher speed limits

Table 12: Stakeholders' needs in hypersonic initiatives: CUSTOMERS

Stakeholders' category: CUSTOMERS	
Stakeholders' identification	**Stakeholders' expectations**
Passengers	• To experience a period of microgravity • To spent a unique experience • To view the earth curvature • To shorten the travelling time
Astronauts	• To be trained in a realistic environment • To experience microgravity • To carry out scientific experiments.

Table 13: Stakeholders identification and needs analysis for the case study.

Stakeholder category	***Stakeholders' identification***	***Stakeholders' needs***
Sponsors	Malaysian private enterprise	• To develop, produce and commercialize a transportation system able to perform parabolic flights. • To provide a routine flight service of parabolic flight with an ad-hoc transportation system. • To demonstrate national capabilities
Operators	Italian and Malaysian companies	• To provide a routine flight service of parabolic flight with an ad-hoc transportation system.
End-Users	• Malaysian private enterprise • Malaysian enterprise • Malaysian ministry • Scientific community	• To receive, analyse and exploit data coming from the mission or from scientific experiments carried out. • To enhance public consensus about commercial flight initiatives.
Customers	Passengers	• To experience microgravity. • To experience an amazing view of the Earth. • To carry out scientific experiments.

Considering the information stored within these tables, some considerations can be derived. In particular, it is possible to highlight that:

- Some needs and expectations are shared among different categories of stakeholders
- Some stakeholders may belong to different categories. Depending on the role under examination, different needs can be envisaged.

- Private enterprises are even more important, especially in the rush for the space commercialization.

An example of Stakeholder Analysis applied to a specific case study is the reported in Table 13 and in depth described in (Fusaro, 2015), (Fusaro, 2017). It refers to a study carried out in a cooperation among Politecnico di Torino, Altec and TAS-I Torino addresses a challenging mission envisaged by a group of private Malaysian investors supported by their research and development ministry.

3.3 Impact of the regulatory framework

The impact of a regulatory framework on the design activities of whatever type of product should be taken into account since the very beginning of the design activities. In case of aviation and aerospace products or operations, different regulatory frameworks have been developed in the past and now they can act as guidelines for the designers. Unfortunately, in case of products conceived with disrupting technologies, as the hypersonic transportation systems and missions are, it could happen that specific regulatory framework has not been fixed yet. However, in the specific field of hypersonic, all the major players in regulations affairs, are currently involved in the challenging activity of defining a new ad-hoc regulatory framework but is a very hard task, especially considering that a world-scale regulation would be the only suitable, at the end. Indeed, as soon as the technologies for this new generation of vehicle have been developed, their legal status became a matter of debate. The first issue to be clarified is whether these transportation systems, especially those one able to perform conventional take-off and landing manoeuvres, should be subject to international air law or the outer space law. This debate is also taking place at national level, in order to understand whether commercial aerospace adventures should be regulated by existing aviation laws or should they fall under new laws specifically designed for this new type of space vehicles. Moreover, it has to be considered that the problems related with regulations are not only affecting the vehicle or the mission in which the vehicle is expected to operate, but also the on-ground infrastructures supporting the vehicle itself either when it is on-ground and all along its mission.

Due to increasing rate of development of suborbital flight initiatives, and considering that a suitable regulation for this kind of vehicles may be easily extended for future point to point vehicles too, the major players in the field of regulation are currently focusing all their efforts in the development of regulatory framework for suborbital vehicles, easily extendable to P2P vehicles.

Considering the trajectories carried out by spaceplanes such as Virgin Galactic's SS2 or the XCOR's Lynx, traversing the entire airspace on the way to (and back) the target altitude, usually fixed at a maximum of about 100 km, the most important legal issue to be solved is whether these vehicles should be governed by air law, by space law or by a hybrid law. Taking a look at the currently initiatives in this framework, two different theoretical approaches may be identified (Jakhu, 2011)

- The *spatialist* approach, aiming at identifying the border or outer space on the basis of the natural properties of the space. Different possibilities have been envisaged:
    1. Outer space begins where there is no longer atmosphere.
    2. Outer space begins where the Earth's gravitational pull is balanced by that of another celestial body (Lagrange points).
    3. Outer space begins where the lowest possible orbital perigee for a satellite can be identified.
    4. Outer space begins where the ability of a state to exercise sovereignty ends.
- The f*unctionalist* approach, aiming at classifying an object as space object on the basis of its function as such object as opposed to an aircraft. This approach would apply air law to those objects engaged in aviation and apply space law to those objects intended to operate in space. This test would apply regardless of where the object happened to be physically located at the time of a legal claim. Considering the case of spaceplanes, the functionalist' theory seems to be more ambiguous than the spatialists' one considering that these vehicles are envisaged to operate as aircraft while they are in atmosphere and as space vehicles where they should enter outer space.

In this context, the International Civil Aviation Organization (ICAO) started to work a lot on this topic, being a specialized United Nations (UN) agency concerned with international civil aviation. Even if suborbital vehicles can hardly be defined as "aircraft", following the definition of the Chicago Convention that established ICAO, it can be said that ICAO still has the jurisdiction over suborbital spaceflight to the degree necessary to ensure the safety of international civil aviation. In case the current studies will demonstrate that suborbital flights would fall within the ICAO jurisdiction, the ICAO would have the authority to issue Standards and Recommendations Practices (SARPs) to be implemented by member states.

The rhymes proposed by Yuri Fattha of the ICAO Space Learning Group, perfectly summarized the guidelines for the development of a good regulation.

I learnt that regulation

Is one of the most difficult things to balance.

It cannot be too late

Cannot be too early

Cannot be overprotective

Cannot be too vague

Cannot be prescriptive

Cannot be too complicated…

Yuri Fattha of the ICAO Space Learning Group

Moreover, due to the importance of this debate and the urgency of generating at least a first draft version of regulations for spaceplanes and, in particular for suborbital vehicles, several States undertook many efforts to develop domestic regulatory regimes. Although the United States was the first Nation to issue a regulation on suborbital initiatives, other states, especially European ones, are currently involved in the generation of a proper code law. In some cases, like Italy is doing through its proper agency ENAC, the different national authorities are basing their studies and proposal on the laws already developed in the US in order to have a first reference draft document and also to be able to host and operate spaceplanes currently developed and operated in the American territory. This approach would shorten the spaceplane and spaceports certification processes. However, at European level, it should be mentioned that the European Aviation

Safety Agency (EASA) explored the creation of harmonized regulations to apply across the European Union (EU) but this attempt should be considered and evaluated together with each member state research and development activities.

Different Memorandum of Understanding (MoU) have been signed in between different parties both on the site of spaceplane developers and producers and future spaceports developers and operators.

Traditionally the USA have been very active in this field and a number of Spaceports have been licensed by the FAA, while different initiatives are going on to develop spaceflight systems that will eventually support commercial market (Santoro, 2016). Several market studies are available in literature, and in particular, the one carried out by the Tauri Group (Tauri, 2012) can be considered a very useful reference. It is a study jointly funded by the Federal Aviation Administration Office of Commercial Space Transportation and Space Florida and shows a worldwide market forecast and it is a very useful starting point for more countries-tailored studies. Another relevant study has been presented in Florida Spaceport System Plan (April 2013). It starts from the analysis of existing Spaceports and addresses future Spaceport Vision envisaging possible relevant implementation.

Among the European Countries, UK has been quite active and the Civil Aviation Authority (CAA) conducted a detailed review of what would be required from an operational and regulatory perspective to enable spaceplanes to operate from the UK by 2018 (CAA, 2018).

3.3.2 The Italian Case Study: the focus on spaceports.

Taking a closer look to the Italian territory, the favourable geographic location in Europe, surrounded by Mediterranean Sea and the climatic condition, as well as the deep aerospace capabilities and skills can play a significant role in considering Italy as a very luring place to setup initiatives of commercial suborbital spaceflight for space tourists but also for microgravity experimentation (Santoro, 2015).

Unfortunately, Italy is currently lacking of a regulatory framework that allows suborbital operations in the country, but activities are on-going following the signature of a Memorandum of Cooperation between the FAA/AST, ENAC and

ASI, to fill the gap basing upon the FAA/AST approach. Once possible Spaceport Locations are identified, the selected site shall need to be licensed by the Italian Civil Aviation Authority. The initial activity to be developed is the identification of a candidate Spaceport in the Country, equipped with all the infrastructures and capabilities needed to adequately support spaceplanes operations. Starting with the assumption to focus on already existing airports and not building a new one from scratch the main requirements for an airport to be eventually considered and licensed as Spaceport shall be elicited. In particular, special emphasis will be given on the Safety aspects that drive many of the selection criteria. In this context, the author has been involved, together with Politecnico di Bari and Altec in the definition of a suitable methodology to help out in the Spaceports selection criteria by evaluating the specific sites of interest as summarized in (Santoro, 2016). Then, in order to overcome the lack of regulations on the Italian territory, suitable approach to the implementation of a licensing process for an Italian commercial spaceport adopting the FAA/AST procedure has also been investigated. In any case, it is clear that specific trajectory simulations should be performed in support to this investigations, referring to a specific Spaceplane concept and Spaceport location.

Eventually, it is important noticing that in general, the term Spaceport designates a site featured with all the technical, infrastructural, operating, safety and the relevant license from the local Authority to allow take-off, landing and ground/flight operations for spaceplanes suitable for suborbital parabolic or point to point flight. Also, since many of the above referred vehicles will be considered reusable in perspective, a Spaceport shall allow all the refurbishment and maintenance operations on the vehicle. For parabolic missions, the departing Spaceport is the same as the landing one, while for the point to point missions the arrival Spaceport is located in a different point of the Earth. The concept of Spaceport does not necessarily imply the building of a brand new facility, but at least initially the exploitation of already existing infrastructures will be considered, may be with the implementation of additional functionalities. This is in particular the case of Italy, considering the touristic vocation of the Country, the geographical location, climatic conditions, particularly suitable to space tourism are such as some sites both civil and military are believed favourable to the purpose.

3.4 Roadmapping activities in support to strategic decisions

The increasing competition in the field of commercial space flight and of course, of hypersonic transportation, brought technology and innovation management to the centre of decision-making processes, giving even greater importance to Strategic Decisions that could no more avoid to rely on rational processes. Indeed, even if these high-level decisions are usually related to non-measurable parameters or to political and economic situations, they must include technological considerations for the development of innovative solutions, to have a chance of being at least sustainable in such a changing and competitive market.

A useful tool to monitor the current technological state and the plans for its future advancement is a technology roadmap. A technology roadmap can be considered the output of the technology roadmapping process, a particular kind of activity flow aimed at identifying and selecting technologies, mission concepts, capabilities and building blocks according to specific strategic plans. Considering a set of targets to reach, it clearly identifies critical system requirements, the product and process performance targets, the technology alternatives and milestones to be pursued. A roadmapping activity can be easily described as a complex process considering many parameters at the same time. For example, a technology roadmap definition process has to consider at the same time current or changing limitation of financial resources by both the government and industry, with scientific or technical needs and with current general public requests. In order to correctly suggest a TRL increase path, financial limitations and stakeholders' needs have to be considered: to this purpose a prioritization of the lists of identified bricks is required to consider them with the right priority. Strategic decision makers need a method to assist them in the prioritization of advanced technological investment.

The author of this thesis is participating to several on-going initiatives in the fields of roadmapping activities initiated and actively supported by the European Space Agency (ESA). In particular, taking advantage of the first research activities carried out in the field of space exploration, (Cresto, 2016), (Cresto, 2017), (Viscio, 2013), (Viscio, 2014) Politecnico di Torino is currently supporting the elaboration of hypersonic and re-entry space transportation systems roadmap

(Cresto, 2017), (Fusaro, 2017b). This research activity foresees the development of a logical methodology based on the combination of common System Engineering tools and processes (such as Functional Analysis, Concept of Operation definition and Decision Analysis), and thus with the possibility of being fully integrated within the complex design methodology suggested in this thesis. It has to be underlined that integration is not only purely theoretical, but as published in (Cresto, 2017) and (Fusaro, 2017b) the methodology has been implemented in a MBSE environment, with ad-hoc developed tools to derive, track and manage the four basic elements of a Technology Roadmap and their features, i.e. Operational Capabilities, Technology Areas, Building Blocks, and Mission Concepts.

Undeniably, even if Europe has access to space, it has a limited experience associated with hypersonic, (re-)entry and landing vehicles on Earth and other celestial bodies with an atmosphere. Despite all these efforts, for Europe there is a urgent need of planning also to increase its presence in the market related to the field of hypersonic and re-entry space transportation systems. In case the reader would be interested in such high-level design aspects, the above mentioned references can be used to in-depth look at the methodology suggested to drive the generation and update of technology roadmaps for hypersonic and re-entry systems. In addition to the methodology, this reference foresees also the development of a flexible and easily updatable database for hypersonic transportation and re-entry systems, strictly connected to the methodology and based on the same four basic elements. HyDat (Hypersonic Database) (Fusaro, 2017b) is currently under development and there is an increasing growth in the attention payed by several major players in the possibility of contributing to this project, envisaging short and long term benefit of the possible exploitation of such a tool for supporting Strategic Decision but also to increase the confidence level of the high-level estimations, usually carried out during the conceptual and preliminary design phases.

Eventually, it is important noticing that the capability of guaranteeing a traceability or even of influencing of the high-level strategic decisions within a unique design methodology, is a giant leap towards the possibility of designing of more competitive aircraft, fully in line with the aim of this work.

3.5 Mission Level Analysis

3.5.1 Process description

3.5.1.1 Mission Statement, Mission Objectives and Mission and Programmatic Requirements – (Theoretical approach)

The results of the in-depth analyses of the stakeholders and relative needs, of the current aerospace market, considering possible limitations imposed by the under-development regulatory framework and the high level strategic decisions, allows to rationalize Mission Statement derivation process as well as the following generation of a first list of Mission Objectives.

Both Mission Statement and Mission Objectives, together with their heritage in terms of related high level needs and constraints from which they have been derived, should constitute the starting point for the elicitation of a first list of requirements usually referred to as Mission Requirements. It is convenient to notice that, in case strategic decisions are considered, programmatic requirements can be generated too. While mission requirements aim at describing the mission and are currently derived by the mission objectives and so, ultimately, from the stakeholders needs, programmatic requirement are usually set by the Mission Directorate, program, project. These include strategic scientific and exploration issues, systems performance, schedule, cost and similar non-technical constraints.

In the following subsection, the mission statement and the mission objectives are derived for the specific case study of the suborbital vehicle aimed at performing parabolic flight services, already presented in the section of the Stakeholder analysis. In the following sections as well as in all the other subsections aimed at reporting the results of the application of the methodology to a specific case-study, both a document-based Systems Engineering approach as well as a Model Based Systems Engineering approach are presented. This strategy will guarantee several benefits:

- To allow readers with limited knowledge in MBSE approaches and SysML to understand the case study.
- The implementation of the selected reference case study may act as a tutorial for readers who are approaching MBSE domain.

- To allow skilled readers to appreciate the major advantages of moving from a Document-based to a Model-based approach in the field of advanced Aerospace Engineering.
- To show to both skilled and non-skilled readers the major benefits of an integrated multidisciplinary design approach since the very beginning of the design process.

3.5.1.2 Mission Statement, Mission Objectives and Mission and Programmatic Requirements – (Case Study)

Continuing the example of the suborbital mission presented in the Stakeholder Analysis section, it is possible to notice the way in which the different elements of the several preliminary analyses (Stakeholders, Market, Regulations and Strategic decision) can contribute to the derivation of the mission statement. As a recap, the selected reference case study deals with the conceptual design of a spacecraft aimed at parabolic flights with the special capability of being able to perform a vertical take-off and landing (VTOL).

In the following mission statement, i.e. a concise and precise phrase, describing the objectives of the mission, the Mission Objectives, i.e. its primary and secondary goals as well as a first draft list of requirements are reported.

Mission Statement:

"The mission shall allow regular flight services to enable 4 flight participants at a time to reach 100 km to experience a period of microgravity and an amazing view of the Earth. The spacecraft shall perform a vertical take-off from a sea-based or land-based platform and a vertical landing on the same site. Moreover, the additional capability to perform an un-crewed mission shall be considered"

Primary Objective:

- *To allow regular suborbital parabolic flights service*

Secondary Objectives:

- *To demonstrate the Malaysian capabilities to develop, produce and operate suborbital vehicles.*
- *To demonstrate the Malaysian capabilities to support regular spaceflight activities.*

- *To demonstrate the possibility of performing parabolic flight with fully reusable transportation systems.*
- *To enhance the public consensus in commercial flight activities*
- *To enhance key-technologies' Technology Readiness Levels (TRLs).*

Mission Requirements:

MR 1. The mission shall allow regular parabolic suborbital flight service
MR 2. The mission shall allow to 4 passengers at a time to experience at least 2 minutes of microgravity
MR 3. The mission shall allow the passengers to reach a flight altitude of at least 100 k of altitude
MR 4. The mission shall allow the passengers to appreciate the Earth curvature
MR 5. The mission shall enable the flight service to be carried out from both land-based or sea-based platforms.
MR 6. The mission shall be conceived in order to guarantee the coincidence of departures and landing sites.

Programmatic Requirements:

PR 1. The mission shall be carried-out on the Malaysian territory
PR 2. The maiden flight shall be performed by the end of 2020
PR 3. The mission shall rely on high TRL technologies as much as possible
PR 4. The mission shall increase the Malaysian role in spaceflight.

In order to move from a pure Document Based to a Model Based Systems Engineering approach, stakeholders and mission objectives have been linked together within a Use Case Diagram (UCD) (Figure 48), following SysML. It can be seen as a graphical representation of a user's interaction with the system that shows the relationship between the users (in this case the stakeholders) and the different use cases (Mission Objectives) in which the actors are involved. The exploitation of a proper layout allows representing and communicating the type of relationships existing among the different elements of the diagram. In particular, in order to express the stakeholder categorizations, generalization links have been used, while to express the interest of each single stakeholder in one or more mission objectives, the association link is suggested. It is also possible noticing that the links allow defining hierarchical relationships between elements; e.g. the secondary objectives are related to the primary one by means of dependency links, with a specific stereotype (“include”). This MBSE approach allows to clearly define the relationships with the Mission Objectives, that are the main bricks of the mission that is going to be designed in the next steps and each of the actors that could be interested in some way to the product or to the services that would derive by its exploitation. These links have not only a pure graphical value but being implemented on a software tool (such as Rhapsody, in this case), they allow to start tracing back to the initiators all the decisions that would be taken all along the design process. These connections between stakeholders and Mission Objectives, i.e. between actors and use-cases represent the very first ring of a chain that would allow at the end of the design process to remember why certain design parameters will have that value or the reason why some alternatives have been selected and other discarded. All starts at this level.

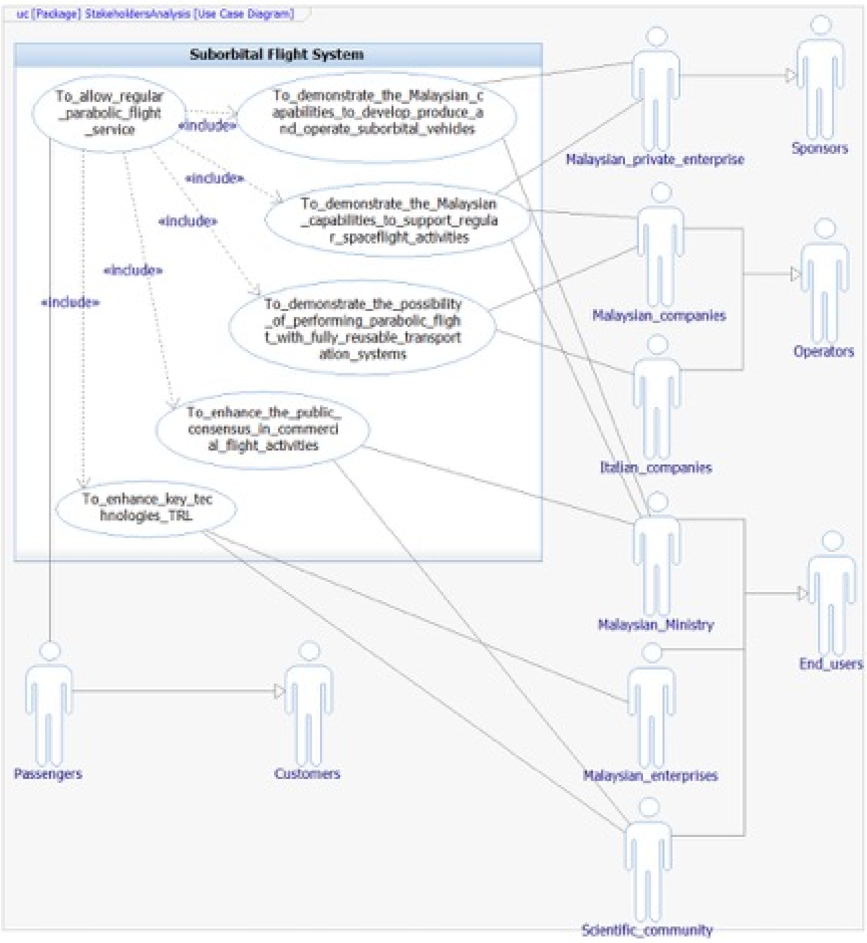


Figure 48: Use Case Diagram for the reference case study.

ID	
MR1	The mission shall allow regular parabolic suborbital flight service.
MR2	The mission shall allow to 4 passengers at a time to experience at least 2 minutes of microgravity.
MR3	The mission shall allow the passengers to reach a flight altitude of at least 100 km.
MR4	The mission shall allow the passengers to appreciate the Earth curvature.
MR5	The mission shall enable the flight service to be carried out from both land-based or sea-based platforms.
MR6	The mission shall be conceived in order to guarantee the coincidence of departures and landing sites.

Figure 49: Mission Requirements implemented in IBM DOORS®.

ID	
PR1	The mission shall be carried-out on the Malaysian territory.
PR2	The maiden flight shall be performed by the end of 2020.
PR3	The mission shall rely as much as possible on high TRL technologies.
PR4	The mission shall increase the Malaysian role in spaceflight.

Figure 50: Programmatic Requirements implemented in IBM DOORS®.

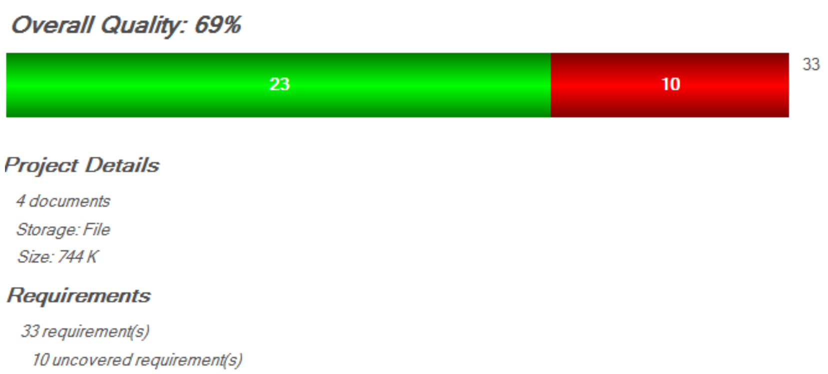


Figure 51: Example of requirements coverage analysis output.

Figure 49 and Figure 50 represent two examples of implementation of different requirements categories in a requirements database. The exploitation of a software based requirements management tool will guarantee several benefits to the design process, since this high-level stages:

- the database allows establishing proper hierarchical levels among requirements guaranteeing the so called internal traceability.
- the database allows linking requirements with Use Cases, i.e. with mission objectives. This is an example of external traceability.
- the database, properly connected with the software allowing the functional and behavioural modelling (such as Rhapsody®) can allow managing the status of the project, giving feedbacks, for example, about the coverage of requirements to the element of the model (see Figure 51)

3.6 Mission Concept alternatives generation, trade-offs and selection.

3.6.1 Process description

Once the main objectives of the mission under investigation have been clarified, the developers should elaborate different ideas to accomplish this mission in the optimal way. Nowadays, the fast technological evolution and the even higher computational capabilities can allow taking into account and manage a very high number of options. One of the main benefits of these innovations is the possibility of postponing the trade-off later-on in the projects, when more accurate data could be available. This subsection provides suggestions on how to manage the very first brainstorming activities, supporting the generation of mission concept alternatives. Starting from a functional view of the mission that allows identifying the different capabilities that the elements of the System of Systems should guarantee, looking at the existing reference missions but also taking a look to possible future near time evolutions, the developers should identify all the possible elements able to accomplish the previously defined functionalities. In order to carry out these two steps, traditional Systems Engineering tools, typical tools developed and used for the Functional Analysis, can be exploited. In particular, the Functional Tree can allow defining the main functions the mission shall perform and a Function/Product Matrix could help to structurally define the variety of elements able to accomplish the previously deduced functions.

3.6.2 Mission Alternatives derivation

This section aims at providing the readers with the knowledge and the tools to derive all the elements and elements' combinations to generate the highest possible number of mission alternatives, following a defined and formalized process, based on a Systems Engineering approach. In particular, in order to accomplish this task, the tools presented in the previous section are exploited and organized in a new way (Figure 52).

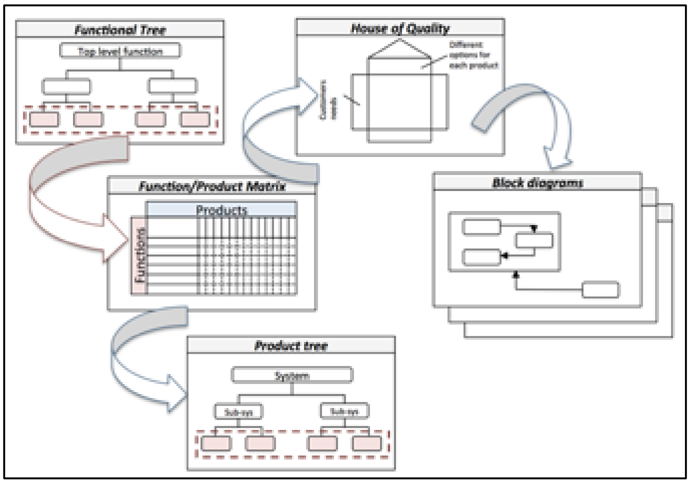


Figure 52: Sketch of the tools of the functional analysis that can be used to derive mission elements.

This is the sequence of suggested activities:

1. Identification of the functionalities required to accomplish the already defined mission objectives. This can be carried out exploiting a traditional functional tree that can be formalized by means of a Block Definition Diagram (BDD) following the MBSE approach. In addition, the first list of functional requirements can be elicited. However, from the grammatical point of view, the subject of these statements cannot be specified, but more

generic nouns shall be exploited. From the end of the next step, a revision process of these requirements will be performed allowing better specifying them depending on the proposed allocation of functions to products. Please notice that the exploitation of Requirements Management tools guarantees to trace all these changes, allowing the engineers, at any time, to verify the evolution of each single requirement.

2. Identification of all the possible products able to perform each single function previously identified. This analysis can be supported by the exploitation of function/product matrix, in a non-orthodox way. Indeed, the usual procedure prescribes that each identified product can be able to perform more than one function, but each function shall be carried out by a single product only. This guarantees an optimization of the resources and allows preventing the user to mix together different hierarchical levels. However, in this context, a non-orthodox exploitation of this tool is suggested, proposing the users to identify and list all the possible elements able to perform each single function. This will result in a matrix with a higher number of valid intersections.

3. Before moving to a pure physical view, it is necessary to assemble mission scenarios through proper combination of one alternative per function. In this context, the exploitation of the Quality Functional Deployment (QFD) tool is suggested. Besides the fact that this tool is not one of the traditional tools of the Systems Engineering, the here proposed exploitation of QFD tool can be suggested as additional tool of a MBSE approach. Notwithstanding, the presented application to the reference case-study will demonstrate that it is possible to fully integrate QFD in MBSE tool chain. The QFD will be exploited within an iterative and recursive process allowing not only the generation of mission scenario alternatives but also their prioritization on the basis of proper criteria, directly coming from the stakeholder analysis.

4. The most promising scenarios, whose number depends on the possibility to carry on parallel analyses for more alternatives, can be furtherly detailed from both a physical and a behavioural standpoint. As far as the physical description is concerned, product tree can be exploited. This is another activity that can be formalized by means of a BDD in SysML. The product tree is here conceived in order to have three hierarchical levels, in order to

be consistent with the level of detail expressed in the functional tree. The suborbital flight System of Systems is the main assembly, whilst three segment-level products have been identified, each of which may consist of other systems. Requirements definition and classification follow this breakdown too.

5. From a behavioural standpoint, the so called Concept of Operation, can be assessed by means of
   i. Block Diagrams showing the connections of the elements (that could be formalized by means of an Internal Block Diagram)
   ii. Functional Flow Block Diagram (FFBD) that allows to describe the right sequence of functions to be performed by the system in order to achieve the mission objective. FFBDs can be formalized by means of Activity Diagram.
   iii. The timeline to be accomplished during the mission that can be formalized by means of Sequence Diagrams.
   iv. The description of the Modes of Operations, in the several mission phases exploiting State Machine (SM) diagrams.

3.6.3 Functional Analysis: process and support tool-chain (Step 1 – Step 2)

3.6.3.1 Functional Tree and Functional Requirements (Theoretical approach)

A Functional Tree expresses the functions to be performed for the execution of the mission. The functional tree allows splitting the highest level functions, which stem from the mission objectives, into lower level functions, through a typical breakdown process, eventually allowing the identification of the basic functions that have to be performed by the identified product. Therefore, starting from the so-called top-level functions, the functional tree generates various branches, moving from the most complex functions to the basic functions, i.e. those functions at the bottom of the tree that cannot be split any further. Starting from the already identified functions, it is possible to derive a first list of functional requirements, in which the subject of each statement is not detailed yet, but it is inherited directly from the level of the analysis (Segment, System, Subsystem, etc…). Then, after the exploitation of a function/device matrix, the first draft list of functional requirements can be updated with the most adapt subject, i.e. the product selected to perform the function the requirement refers to. This is a clear example of changes that may affect the list of requirements during the overall life cycle process. It is not easy to track, save and manage all these changes, especially with the level of detail increases. The exploitation of a proper requirements management tool, in this case the IBM Doors®, can guarantee this capability.

3.6.3.2 Functional Tree and Functional Requirements (Case study)

Figure 53 shows the Functional Tree for the reference case study. It is possible noticing that starting from the so-called top level function (i.e. To perform regular suborbital flight) three segment-level functions have been derived. Each of these functions express some capabilities requested to the overall mission and only one of them (i.e. To transport passengers) seems to be more related to the air transportation system. However, looking at the overall mission, it is very important not to neglect the other functions at least at this design stage. Then, as it will be clearly shown in the next Chapters, aiming at designing the aircraft, only the relative function will be furtherly decomposed up to the desired level of detail. The other functions will be decomposed only in case important interfaces should be designed or taken into account.

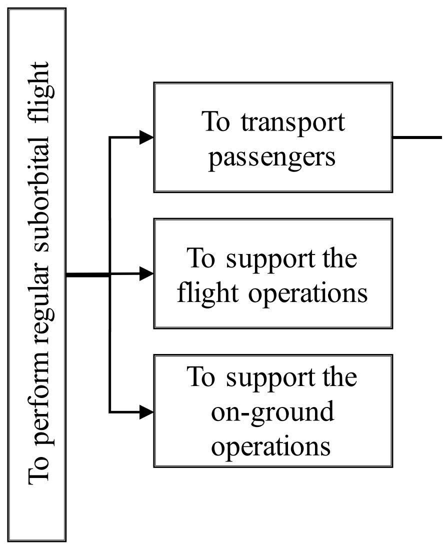


Figure 53: Functional Tree for the reference Case Study, stopped at Segment Level

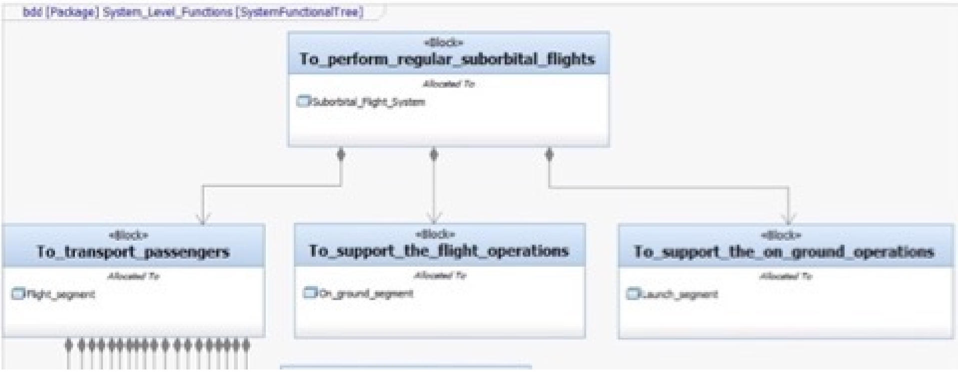


Figure 54: BDD implementing a Functional Tree for the reference Case Study, stopped at Segment Level

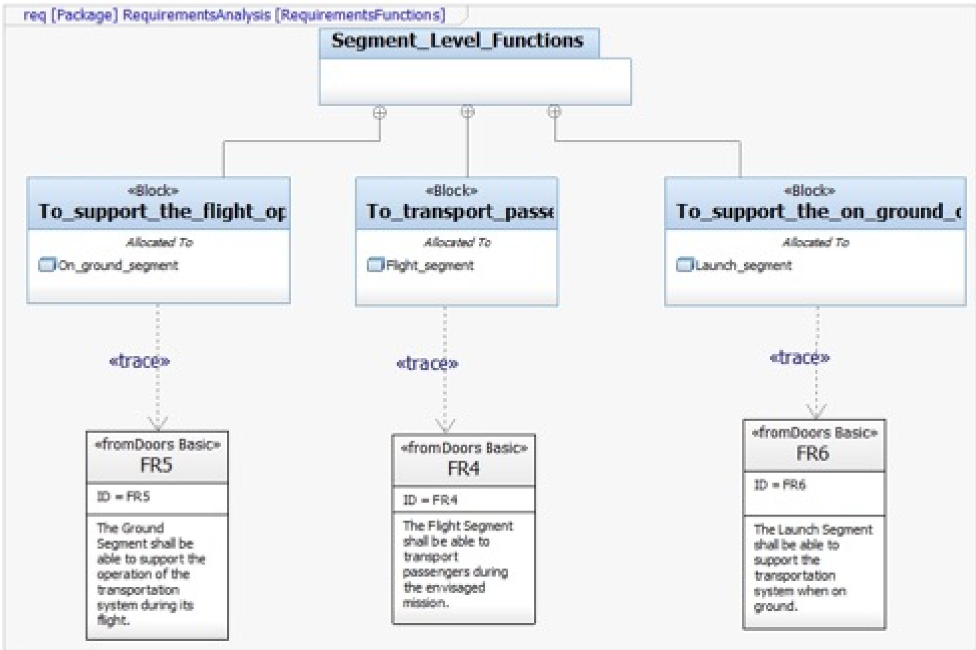


Figure 55: BDD implementing a Functional Tree for the reference Case Study, stopped at Segment Level with links to Functional Requirements

Figure 54 depicts a the implementation of the functional tree reported in Figure 53 by means of Block Definition Diagram (BDD) in which each single block represents a specific function. Besides the already mentioned and discussed advantages in terms of hierarchical representation, this Figure shows another benefit of the MBSE approach. Indeed, each block contains also the information of the physical element of the mission on which it is allocated. Please, take care this is the result of the following two steps of the methodology that are discussed immediately after this section. In addition, once the Functional Requirements have been elicited, stored in the requirement databased and allocated to the relative functions, it is also possible to have some alternative views of the Functional Tree with additional very useful information .(see Figure 55)

3.6.3.3 Function/Product Matrix and functional requirements refinements (Theoretical approach)

It allows identifying the elements or building blocks needed to accomplish the functions. Specifically, the matrix's rows contain the basic functions coming from the functional tree, while the columns report the products, i.e. the space mission elements capable of performing those functions. Starting from the analysis of the first basic functions, new elements progressively fill in the columns. Eventually, all basic products are determined. As a result, the elements to be involved in the missions are identified, by mapping all basic functions to products. As anticipated, the functional requirements list can be refined, substituting the generic subjects with proper product names. It is important to notice that the generation of the function/product matrix, as well as well as the requirements list, should follow the same hierarchical organization already expressed and formalized in outlining the functional tree.

Besides general SE rules suggests not to identify more than a product able to guarantee a certain function, at this level of design, the functional analysis shall also be used to derive alternatives and thus, the more the products able to carry out a function will be, the highest the number of mission alternative scenarios and thus, highest the possibility of enhancing the level of innovation of both the mission and the product, widening the design space.

3.6.3.4 Function/Product Matrix and functional requirements refinements (Case Study)

Taking a look to the reference case-study, as it has been stated in the previous section, the matrix is exploited to map all the possible elements of the mission able to carry out the three identified segment level functions.

In this case the MBSE approach allows to express this subtle variation with respect to the nominal case exploiting a different link to connect functions and products. In particular, it can be noticed that even if the generic name of the segment can be associated uniquely to the function exploiting allocation links (see Figure 57), going in the details of the elements, different alternatives may be possible and thus generalization link may be exploited (see Figure 58).

		Flight Segment Alternatives					Ground Segment Alternatives				Launch Segment Alternatives					
		Capsule	Lifting Body	Re-entry System	Missile	Spaceplane	Existing sea-based Infrastructure	Existing land based Infrastructure	New sea-based Infrastructure	New land-based Infrastructure	None	Rocket	Helicopter	Balloons	Airship	Aircraft
Functions	To transport passengers															
	To support the flight operations															
	To support the on-ground operations															

Figure 56: Function/Product Matrix at Segment Level

	Flight_segment	Launch_segment	On_ground_segment
To_transport_passengers	Flight_segment		
To_support_the_flight_operations			On_ground_segment
To_support_the_on_ground_operations		Launch_segment	

Figure 57: Function/Product Matrix at Segment Level (with allocation links)

	Capsule	Lifting_body	Re_entry_system	Missile	Spaceplane
To_transport_passengers	Capsule	Lifting_body	Re_entry_system	Missile	Spaceplane
To_support_the_flight_operations					
To_support_the_on_ground_operations					

Existing_sea_based_infrastructure	Existing_land_based_infrastructure	New_sea_based_infrastructure	New_land_based_infrastructure
Existing_sea_based_infrastructure	Existing_land_based_infrastructure	New_sea_based_infrastructure	New_land_based_infrastructure

Rocket	Helicopter	Balloons	Airship	Aircraft
Rocket	Helicopter	Balloons	Airship	Aircraft

Figure 58: Function/Product Matrix at Segment Level (with generalization links)

Looking at the case study, it is possible noticing that this method allows the identification of a very high number of design alternatives that would be directly translated in a very high number of Mission Concept alternatives. Moreover, at this stage, the first Functional Requirements may be derived and stored within the database. However, their subject can only be very generic. Once the mission alternative will been selected, these requirements will be refined.

ID	
FR1	**1 Top Level Requirement**
FR2	The Suborbital Flight System shall be able to perform regular suborbital flights.
FR3	**2 Segment Level Requirements**
FR4	The Flight Segment shall be able to transport passengers during the envisaged mission.
FR5	The Ground Segment shall be able to support the operation of the transportation system during its flight.
FR6	The Launch Segment shall be able to support the transportation system when on ground.

Figure 59: Segment-level funcional requirements

	FR2	FR4	FR5	FR6
To_perform_regular_suborbital_flights	FR2			
To_transport_passengers		FR4		
To_support_the_flight_operations			FR5	
To_support_the_on_ground_operations				FR6

Figure 60: Segment-level funcional requirements links with functions

	FR2	FR4	FR5	FR6
Suborbital_Flight_System	FR2			
Flight_segment		FR4		
Launch_segment				FR6
On_ground_segment			FR5	

Figure 61: Segment-level funcional requirements links with mission segments

3.6.3.5 Product Tree and Design Requirements (Theoretical approach)

Product Tree is a functional analysis tool used to represent the product breakdown structure of the system, with a level of detail coherent with the functional analysis performed above. It is mainly for this reason that this tool is strictly related to the Function/Product Matrix. From this diagram, it is possible to define a new list of requirements, usually referred to as Design Requirements, suggesting the way in which the lower-level components can be integrated to form the highest level component.

3.6.3.6 Product Tree and Design Requirements (Case Study)

Figure 62 presents a generic product tree stopped at Segment Level. As it has been previously described, for each of these generic elements categories, different alternatives may be envisaged and these differences (between element selections and elements alternatives) may be easily reproduced exploiting SysML has outlined in Figure 63 and Figure 64.

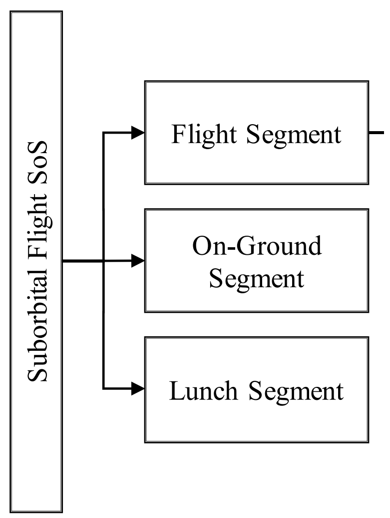


Figure 62: Segment-level Product Tree

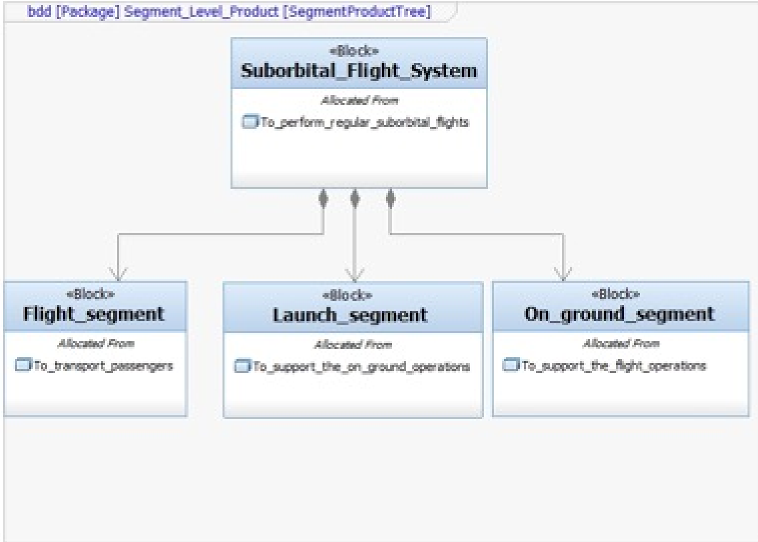


Figure 63: Segment-level Product Tree in MBSE

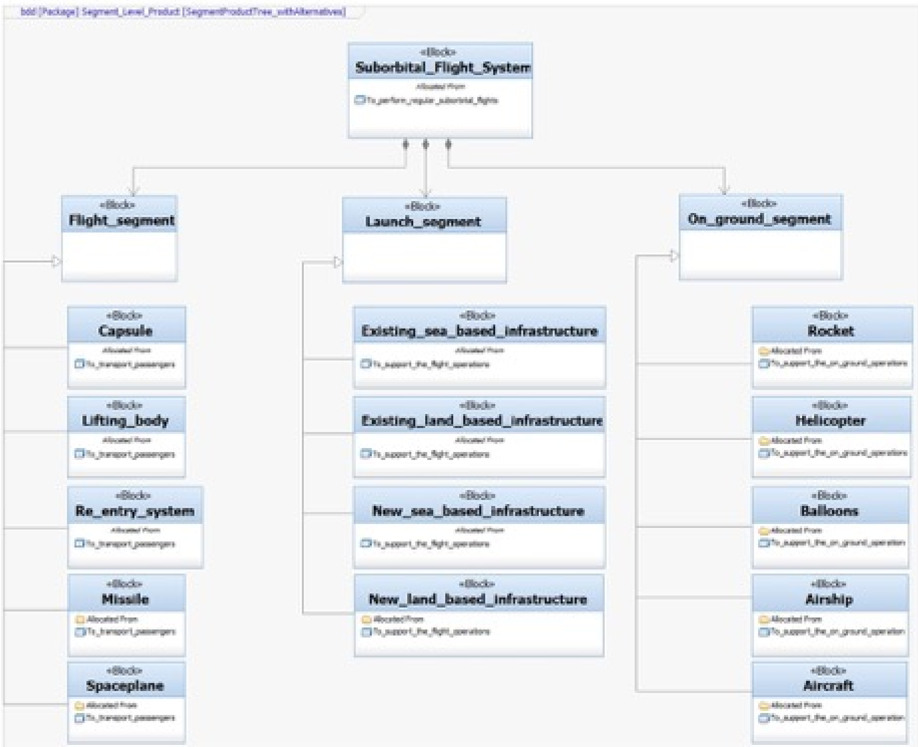


Figure 64: Segment-level Product Tree in MBSE with alternatives

3.6.4 Innovative Quality Functional Development Tool (Step 3)

At this point of the design process, when different alternatives per each major elements of the mission have been identified, it is important to group and combine those elements to derive the different mission concept options. During this process it is fundamental to evaluate how well each of the different option able to solve each single function is able to accomplish the function itself and which is its relation with all the other functions of the mission. In order to increase the level of autonomy of the process, and to allow an integration within a multidisciplinary design methodology, the author suggests to use the Quality Function Deployment (QFD) tool, also known as House of Quality.

The Quality Function Deployment Tool is a very useful design method to transform qualitative user demands into quantitative parameters, to deploy the functions forming quality and to deploy methods to achieve the design quality into subsystems and component parts, and, ultimately, to specific elements of the manufacturing process, as described by Dr. Yoji Akao (Akao, 2003). From its first theorization, this method has been applied in very different domains (Chan, 2002). In particular, it's become widespread exploited in many design applications, not only at top-level, but also at system and sub-system or equipment levels. From the graphical point of view, the QFD tool it's very similar to a sort of house (in fact it is very well known has House of Quality), with external walls, bases and a roof.

The basic House of Quality consists of the following parts:

- Rows definitions.
- Rows weighting factors
- Columns definitions
- Interaction Matrix
- Relationship Matrix
- Scores or prioritization

Furthermore, aiming at integrating this tool within the envisaged tool-chain, additional weighting rules have been added. It is worth to notice that the QFD has not been developed to be used as a stand-alone graphic, but its better exploitation could be obtained within a QFD tool- chain that allows obtaining suggestions for engineering parameters, starting from top-level market analysis.

The usual sequence of QFDs, covering the overall product life cycle is reported in Figure 65. The tool-chain suggested and described in this subsection aims at providing as main output a series of prioritized mission concept options able to satisfy the top-level mission requirements. It's up to the engineers the definition of the number of options to select for the follow-on of the process. Depending on the number of personnel, budget and time schedule, it would be convenient to carry on at least two or three different mission concepts since the following design milestone. This could be a conservative approach, preserving from unexpected changings at geo-political, management or economical levels. Moreover, carrying on the procedure, it is possible to obtain a list of enabling technologies from which it is possible to define development roadmaps (Cresto, 2015), (Cresto, 2016) (Viscio, 2013)

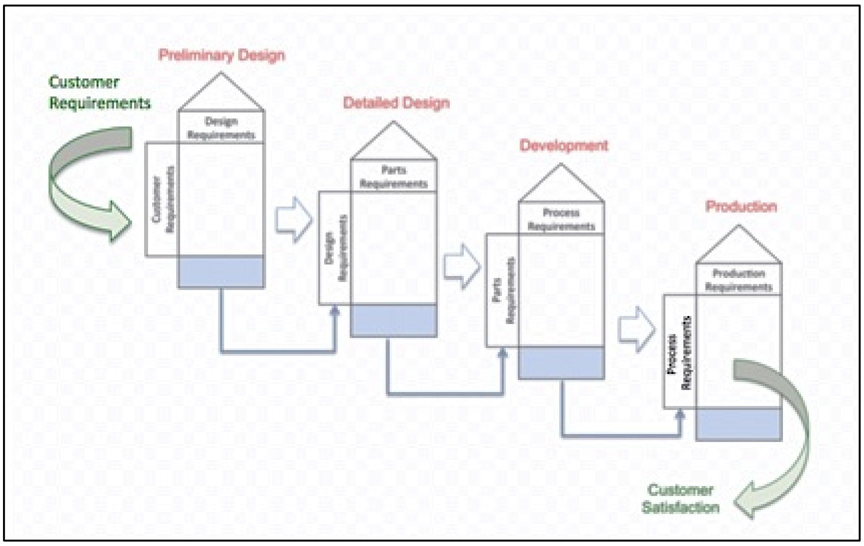


Figure 65: QFD tool chain covering the overall product life-cycle

The first use of the QFD in this methodology aims at discovering the importance of each top-level mission building block in an aerospace mission. In this case, the rows shall contain the list of top-level requirements mainly coming from the stakeholder analysis while columns shall be filled in with the primary building blocks of the mission, obtained by the joint exploitation of Functional Tree and Functions/Products matrix, stopped at the desired level, in this case Segment Level. The scores obtained as output give the designer an overview of the relative importance of each building block for the considered mission. This is very important not only from a pure technical point of view, but also from a managerial perspective. In fact, the building block with the highest score should be in-depth analysed and additional efforts should be devoted to its development, in terms of personnel, resources or budget, because it is the mission component on which the customer requirements will have the major impact.

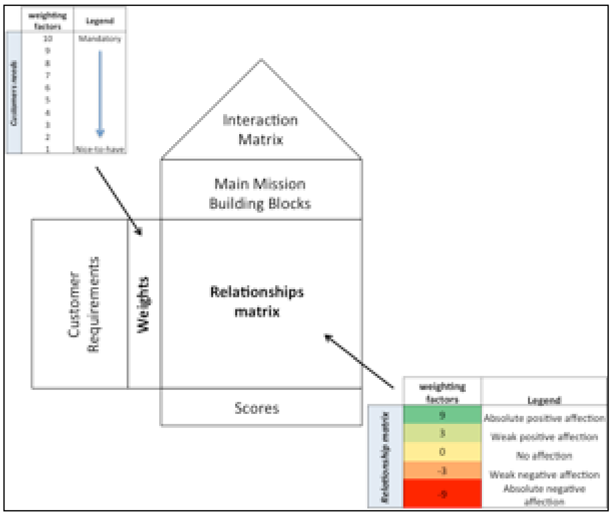


Figure 66: QFD Initialization

The starting point is the requirement weighting process. This activity is a direct consequence of the analyses carried out at the very beginning allowing the elicitation of the first draft list of mission requirements. Depending on the wishes of the stakeholders, the deduced requirements can have different levels of importance. In this context, a classical weighting assignment procedure is suggested to weight the requirements from 1 to 10, where the maximum score is assigned to constraints and the minimum is related to low impact nice-to-have (Figure 66). Besides the foreseeable negligible impact of some requirements, it is useful to take them into account because of their direct impact on some mission elements or on the overall configuration. The same reasoning is also valid for the other top-level requirements coming from other sources such as regulations or geo-political aspects. The following step is the definition of the impact of the building blocks on the requirements satisfaction (i.e.: "How well this element is able to fulfil the requirement?"). Several strategies could be used at this purpose. In this case, a modified version of the classical QFD scoring strategy is suggested, giving the possibility to fill in the matrix Customer needs/Products with:

- "0" in case the requirement is not affecting the product design

- "3" in case the requirement is moderately affecting the product design
- "9" in case the requirement is strongly affecting the product design.
- "-3" in case the requirement is moderately against the product design.
- "-9" in case the requirement is strongly against the product design.

The author suggests an extension of the ranking rules embracing the possibility that a one or more defined mission elements could be in contrast with some of the previously stated requirements.

Moreover, a requirement with a weight greater or equal to 8 cannot admit elements with negative influence score on it. If it happens, the related element should unavoidably be erased from the list of options. Once the scoring process has been concluded, it is possible to rank the elements inserted in the columns. This is obtained applying the following equation:

$$S_{BB_j} = \sum_{i=1}^{n_{req}} \left[\left(w_{req}\right)_i \cdot (w_{rel})_{ij} \right]$$

where:

i is the requirements index;

j is the Building Blocks index;

S_{BB_j} represents the score related to the j-th Building Block;

$\left(w_{req}\right)_i$ is the weighting factor assigned to the i-th requirement.

$(w_{rel})_{ij}$ is the weighting factor assigned within the relation matrix

Then, a second QFD matrix could be used in order to prioritize the mission elements options. Indeed, each building block has to be considered as a collection of interconnected elements. At top level, it is important to consider all the possible options for the elements of a mission. To this purpose, the methodology has been applied to prioritize the mission elements. In order to perform this activity in a logical and structured way, several different QFDs shall be built, one per each original function of the Functional Tree, and use a combination algorithm later on, in order to generate the different mission concept options.

Applying the same above-described methodology, the mission elements prioritization could be obtained applying the following equation:

$$(S_{EO})_{lm} = \sum_{i=1}^{n_{req}} \left[\left(w_{req}\right)_i \cdot (w_{rel})_{il} \right]$$

where:

i is the requirements index;

l is the element options index;

$(S_{EO})_{lm}$ represents the score related to the l-th element option able to accomplish the m-th mission function;

$\left(w_{req}\right)_i$ is the weighting factor assigned to the i-th requirement.

$(w_{rel})_{il}$ is the weighting factor assigned within the relation matrix

The values obtained could be used to prioritize the options for each element. If the process is carried out for each function that the mission shall perform, the engineers can have several rankings, one for each function (Figure 67). The following step implies the combination of the elements in order to create mission concept options (Figure 68 and Figure 69) . This activity can be automatically performed making all the existing combinations, sorting one element per list.

Remembering that each element has been previously scored, the score related to each derived mission concept is a linear combination of the scores obtained in the previous steps, as stated by the following equation:

$$(MC)_k = \sum_{p=1}^{n_{ele}} [(S_{EO})_p]$$

where:

k is the mission concept index;

p is the element options index;

$(S_{EO})_p$ represents the score related to the l-th element option able to accomplish the m-th mission function;

The number of possible combination will be exactly foreseen since the beginning using the following equation:

$$n_{MC} = \prod_{q=1}^{n_{fun}} (n_{eo})_q$$

where

n_{MC} is the maximum number of mission concept options;

n_{eo} is the overall number of element options;

n_{fun} is the number of functions (i.e. the groups from which element options should be taken).

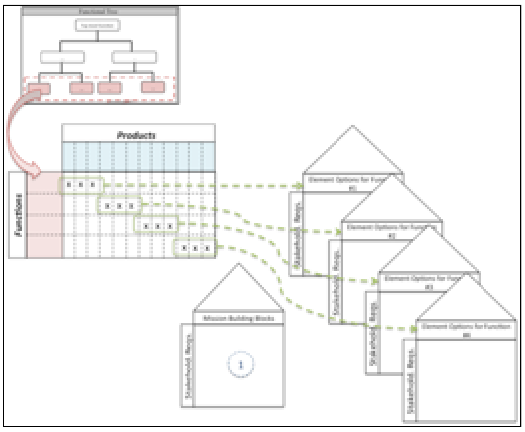


Figure 67: From the functions identification to the mission elements prioritization.

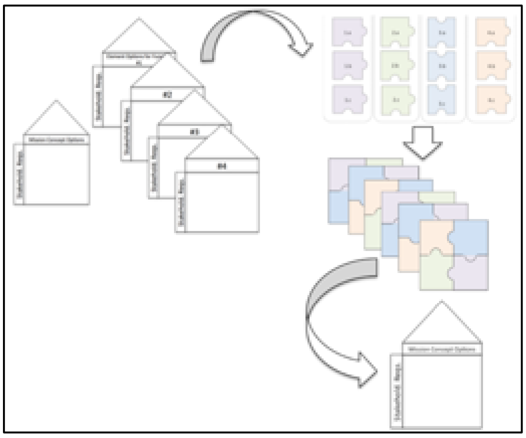


Figure 68: From the mission elements prioritization to the mission concept proposal.

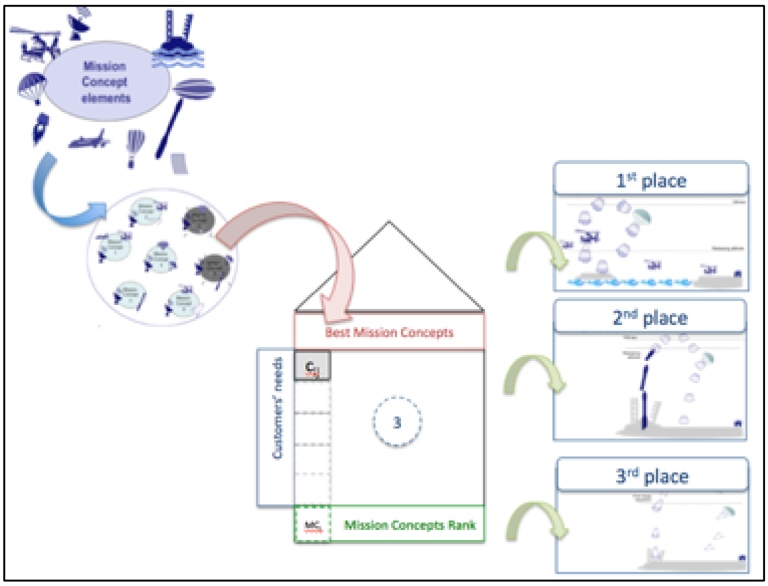


Figure 69: Mission concept options prioritization.

The mission concepts derived exploiting the QFD technique are simple combinations of elements like puzzles. It is clear that an additional characterization is required because a system is not only defined by the elements themselves, but also by their mutual connections. In particular, to discover the relationships among the elements, it is possible to apply again a SE approach revising the results previously obtained. Thus, it is important to start from a functional point of view in order to reach a physical and operative perspective. At this purpose, several tools of the Functional Analysis could be employed. In particular, the Product Tree, Block Diagrams and Functional Flow Block Diagrams are suggested. Please notice that is convenient to apply this and the following steps of the methodology, only at the mission concept options selected as baselines. This precaution can avoid worthless waste of time and money.

3.6.4.1 Innovative Quality Functional Development Tool (Case Study)

The exploitation of Functional Trees and Product Devices Matrixes serves to build the bases for the application of the QFD tool and the whole QFD tool-chain aimed at obtaining the highest possible number of mission concepts, because from the connection matrix it is possible to obtain the columns of the first QFD matrix. Indeed, the rows contain the stakeholder requirements. Figure 70 shows the initialization of the QFD matrix with the selection of the most important stakeholders requirements and with the assignation of the weighting factors.

Subsequently, the QFD has been exploited with the aim of prioritizing the different mission segments. Then, the following Figures show the QFD obtained for the specific case study following the process sketched in this Chapter. The exploitation of this QFD chain allows the definition of the main element options for each identified building blocks. For the sake of clarity, consider that the weighting factors of the relationship matrix have been assigned following the legend in Figure 66. Considering that the design here reported is at the very beginning of the product development cycle, it is important to notice that it is not possible to associate all the parameters with mathematical evaluations, but some of them remain qualitative assumptions. Nevertheless, these assumptions are not so fantastic and will have to be confirmed at later stages of development and analysis. For example, considering that the need of benefitting of a proper view of the Earth has been considered of high importance for the design of the spacecraft but not of extreme importance (indeed, a weight of 6 has been assigned, instead of 9). The main reason for this choice was that level 9 has been assigned only in those cases in which the need is so oppressive that the designers can envisage only one way to carry it out, meaning that this need is impacting and strongly affecting the system design. In this case, you can guarantee a proper view of the Earth in different ways, for example you can enlarge your glass surface (with related structural drawbacks) or exploit innovative technologies like O-LED panels and external cameras, able to make passengers feel an immersion in the external environment (with less structural drawbacks but higher power consumption requirement). For the sake of clarity, each of these reported QFDs is correlated with a Table showing the rational of the weighting or scoring process applied.

				Ground Segment Areas of influence			Flight Segment Areas of influence			Launch Segment areas of influence				
		Weighting Factors	*Normalized Weighting factors*	*Infrastructure*	*Safety*	*Trajectory*	*Staging strategy*	*Payload capability*	*Reusability*	*Layout configuration*	*On-board Systems*	*Vehicle performances*	**Need Priority**	**Normalized Need Priority**
Customer needs	Suborbital mission profile	*10*	*0,108*	9	0	9	9	3	0	0	9	9	480	55,81
	100 km target altitude	*10*	*0,108*	3	0	9	9	9	0	0	9	9	480	56,45
	120 sec microgravity	*8*	*0,086*	0	0	9	3	9	0	3	9	9	336	32,52
	Proper view of the Earth	*6*	*0,065*	0	0	3	0	0	0	3	3	0	54	4,065
	Safe escape system	*10*	*0,108*	0	9	3	9	0	0	9	9	0	390	46,13
	Easy Boarding	*5*	*0,054*	0	3	0	0	0	0	9	3	3	90	5,806
	Accomodation for 4 flight participants	*10*	*0,108*	0	0	0	0	9	0	9	9	0	270	31,94
	VTOL	*10*	*0,108*	3	9	9	3	0	0	3	9	9	450	52,9
	Landing at the take-off location	*10*	*0,108*	9	9	9	0	0	0	9	9	9	540	62,9
	Short time-to-market	*7*	*0,075*	-3	-3	0	0	3	-3	-3	-3	0	-84	-7
	Routine services	*7*	*0,075*	9	3	0	-9	0	9	0	3	9	168	13,77
	Design specification priority			282	285	480	261	303	42	366	645	510		
	Design specification priority (normalized)			3,0323	3,065	5,161	2,806	3,258	0	3,935	6,935	5,484		

Figure 70: QFD initialization for the reference case study

				Ground Segment Alternative					
		Weighting factors	Normalized Weighting factors	Existing sea-based Infrastructure	Existing land-based Infrastructure	New sea-based Infrastructure	New land-based Infrastructure	Impact Area Priority	Normalized Impact Area Priority
Ground Segment Areas of influence	Infrastructure	282	3,032	3	3	-3	-3	0	0,0
	Safety	285	3,065	9	-3	9	-3	3420	36,8
	Trajectory	480	5,161	3	3	3	3	5760	61,9
Launch Segment Areas of influence	Staging strategy	261	2,806	0	0	0	0	0	0,0
	Payload capability	303	3,258	0	0	0	0	0	0,0
	Reusability	42	0,452	0	0	0	0	0	0,0
Flight Segment Areas of influence	Lauyout configuration	366	3,935	-3	-3	0	0	-2196	-23,6
	On-board Systems	645	6,935	-3	-3	0	0	-3870	-41,6
	Vehicle performances	510	5,484	-3	-3	3	3	0	0,0
	Design specification priority			288,0	-3132,0	4689,0	1269,0		
	Design specification priority (normalized)			3,1	-33,7	50,4	13,6		

Figure 71: QFD exploitation for the prioritization of Ground Segment Alternatives

				Flight Segment Alternative						
		Weighting factors	Normalized Weighting factors	Capsule	Lifting Body	Re-entry System	Missile	Spaceplane	Impact Area Priority	Normalized Impact Area Priority
Ground Segment Areas of influence	Infrastructure	282	3,032	-9	-9	-9	3	9	-4230	-45,5
	Safety	285	3,065	-3	3	3	3	9	4275	45,97
	Trajectory	480	5,161	3	3	3	3	3	7200	77,42
Launch Segment Areas of influence	Staging strategy	261	2,806	3	3	3	3	9	5481	58,94
	Payload capability	303	3,258	-3	3	3	9	9	6363	68,42
	Reusability	42	0,452	-9	3	-3	-9	9	-378	-4,06
Flight Segment Areas of influence	Lauyout configuration	366	3,935	0	3	3	3	9	6588	70,84
	On-board Systems	645	6,935	-9	-3	-3	3	9	-1935	-20,8
	Vehicle performances	510	5,484	-9	-3	3	6	9	3060	32,9
	Design specification priority			-12852	-792	2016	12366	25686		
	Design specification priority (normalized)			-138,2	-8,52	21,7	133	276,2		

Figure 72: QFD exploitation for the prioritization of Flight Segment Alternatives

		Weighting factors	Normalized Weighting factors	Launch Segment Alternatives: None	Rocket	Helicopter	Balloons	Airship	Aircraft	Impact Area Priority	Normalized Impact Area Priority
Ground Segment Areas of influence	Infrastructure	##	3,03	9	-9	3	3	3	3	3384	36,4
	Safety	##	3,06	9	-9	-3	-3	-3	3	-1710	-18,4
	Trajectory	##	5,16	0	0	-3	-3	-3	-3	-5760	-61,9
Launch Segment Areas of influence	Staging strategy	##	2,81	0	9	3	3	3	9	7047	75,8
	Payload capability	##	3,26	-9	-3	-9	-3	-3	3	-7272	-78,2
	Reusability	42	0,45	9	-3	3	3	3	9	1008	10,8
Flight Segment Areas of influence	Lauyout configuration	##	3,94	0	0	-3	-3	-3	-9	-6588	-70,8
	On-board Systems	##	6,94	0	0	0	0	0	0	0	0,0
	Vehicle performances	##	5,48	9	3	-3	-3	-3	9	6120	65,8
	Design specification priority			7710	-1893	-4434	-2616	-2616	8844		
	Design specification priority (normalized)			79,0	-24,3	-63,4	-43,8	-43,8	55,8		

Figure 73: QFD exploitation for the prioritization of Launch Segment Alternatives

Table 14: QFD weightings rationale (proper view of the Earth and routine services).

	Proper View of the Earth: *Numerical Value*	Proper View of the Earth: *Rationale*	Routine services: *Numerical Value*	Routine services: *Rationale*
Infrastructure	0	No specific on ground infrastructure should be developed in order to guarantee to the passengers a good view of our planet	9	The need for guaranteeing a routine service has an extreme impact on the infrastructures, as far as maintenance and logistics operations are concerned.
On Ground Safety	0	On ground safety is not affected by this stakeholder need	3	Routine services imply a higher frequency of flights and can impact on the level of risk of incidents on the
Trajectory	3	In order to guarantee a proper view of the Earth, a proper trajectory should be planned, especially as far as the attitude of the spaceplane is concerned	0	The need for routine services does not affect the trajectory definition
Staging Strategy	0	Decision about the staging strategy is not affected by this requirement.	-9	The need for routine services implies a high reusability of the space segment. This appears to be in contradiction with respect to the presence of multi stages.
Payload capability	0	Payload capability is not affected by this requirement	0	The payload capability is not affected by the need for routine services.
Reusability	0	Considerations about reusability of the entire systems or of some of its components is not affected by this requirement	9	On the contrary with respect to the staging strategy, routine services obliges to design the spacecraft guaranteeing reusability.

Layout Configuration	3	This requirement impacts on the transportation system layout, e.g. in the presence of windows, position and sizing.	0	The external layout configuration is not affected specifically involved in a design aimed at guaranteeing routine services
Systems	3	This requirement can affect the internal layout of the crew compartment. Moreover, considering that windows are not the only way to guarantee this requirement, OLED screen and other high-tech on board equipment can be envisaged.	3	Spaceplane systems should be properly designed and installed in order to allow routine services.
Vehicle performances	0	Vehicle performances Payload capability are not affected by this requirement	9	The major performances of the spaceceplane are deeply influenced by this requirement. Please note that among the vehicle performances, key parameters like Turn Around Time.

Table 15: QFD weightings rationale (on-ground operations influence on the aircraft design)

	Layout Configuration		***Systems***		***Vehicle performances***	
	Numerical Value	Rationale	Numerical Value	Rationale	Numerical Value	Rationale
Existing Infrastructures exploitation	0	Layout Configuration is not affected by Operational Constraints	0	The trade off between existing and new infrastructures does not have impact on the design of the systems	3	In case existing on ground infrastructures should be exploited, several constraints for the vehicle performances can arise. In particular, those related to take off and landing phases.
New Infrastructures development	0		0		0	In case new infrastructures will be developed, no specific performance constraint shall be imposed.
Qualified personnel	0		0	The presence of qualified personnel to support on ground ops does not affect the design of the systems	0	The presence of qualified personnel to support on ground ops does not affect the vehicle performances definition

Communication Architecture	0		9	On –board systems design, selection and integration on the spaceplane are deeply affected by the selected communication architecture.	9	Communication architecture can influence vehicle performances.
Sea Based	0		3	The type of location hosting the spaceport can influence the design of certain spaceplane systems.	3	The type of location hosting the spaceport can influence the performances of the spaceplane, especially, those regarding take off, climb, descent and landing. In addition, the vehicle performances in emergency condition can be affected.
Land Based	0		3		3	

Table 16: QFD weightings rationale (on-ground operations influence on the aircraft design)

		Layout Configuration		***Systems***		***Vehicle performances***	
		Value	**Rationale**	**Value**	**Rationale**	**Value**	**Rationale**
Launcher Options	***None***	0	Not having a launcher does not imply any constraint for the layout configuration.	0	At high level, systems are neither positively nor negatively affected by the option envisaged for the launcher	9	If the vehicle shall be able to perform the take-off without a launching system, systems shall be properly sized, especially the propulsion subsystem
	Rocket	0	If a rocket is envisaged as launcher, no special constraints for the layout shall be considered	0		3	The exploitation of a rocket shall moderate impact on some vehicle performances, especially those related to the ascent phase.

Helicopter	-3	The exploitation of helicopter, balloons or a proper airship to lift the spaceplane up to a certain altitude shall influence the layout configuration, imposing a moderate numbers of constraints.	0	-3	The exploitation of helicopter, balloons or a proper airship to lift the spaceplane up to a certain altitude shall influence vehicle performances, especially during the separation phase
Balloons	-3		0	-3	
Airship	-3		0	-3	
Aircraft	-9	The exploitation of mother-ship as launcher imposes the highest number of constraints to the layout configuration	0	9	The spaceplane shall benefit of the presence of a mother-ship, mainly in terms of required thrust and mass of propellant up to be stored.

In the following paragraphs, as example, the first six different mission concepts able to comply with initial stakeholder requirements are described. Each mission concept option is correlated with a brief textual description of the mission and some comments. Please note that these lists are a direct consequence of the scores obtained by the mission concept options in the QFD tool applications.

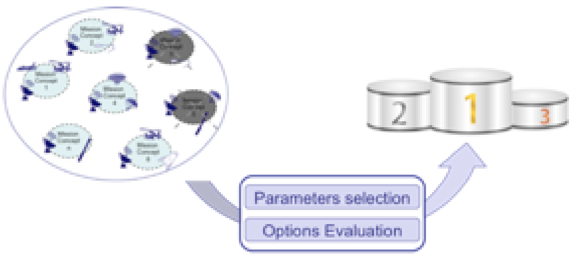


Figure 74: From mission concepts proposal, to the mission concepts selection

Among the hundreds of alternatives of mission concepts arisen from the application of the conceptual design methodology, following the results of the analyses, the last option proposed in this subsection has been selected as baseline. The selection has been carried out evaluating the final ranking (based on the previous QFD matrices) and it is interesting to notice that the numerical suggestions are in accordance with the qualitative comments presented in the previous table. The stakeholders and the developers usually jointly perform this fundamental selection and it is in this special moment that new top level needs or requirements can arise implying a new iteration of the methodology.

Table 17: Mission Concept Alternative #1

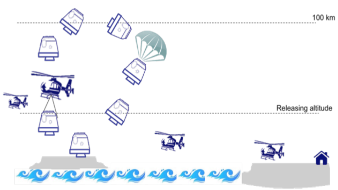
<table>
<tr><td rowspan="4">Helicopter-lifted capsule</td><th>Mission scenario</th><th>Mission description</th><th>Comments</th></tr>
<tr><td>
</td><td>A helicopter coming from a ground-based infrastructure, reaches a sea-based platform and lifts a capsule up to a to be defined (TBD) altitude. Then, after the release, the capsule is appropriately propelled to the target altitude (100 km) allowing flight participants to experience microgravity for few seconds. During the atmospheric re-entry, a series of parachutes should be deployed to slow the capsule down gradually. Additional devices like retro-rockets and control surfaces should be hypothesized to land in the same place from which it takes off.</td><td>This mission concept is characterized by a high level of complexity and risk mainly due to the presence of the helicopter. On the other hand, the capsule-like architecture of the spacecraft guarantees simplicity of the design but a lower level of comfort w.r.t. a spaceplane. Please notice that additional on ground facilities and rescue subsystems should be envisaged. This kind of mission can be suitable for demo missions such as, for example, those aimed at testing and verifying the vertical take off and landing capabilities.</td></tr>
<tr><th colspan="3">Mission Elements Description</th></tr>
<tr><td colspan="3">Ground-based platform: The ground-based infrastructures shall support the operations of a heavy–lift helicopter. In order to minimize the turn around time and for economical reasons, the platform shall be located in a coastal region, in the proximity of a sea-based platform from which the core of the mission is planned to start.
Sea-based platform: The sea-based platform shall host the infrastructures to accommodate and maintain the capsule and support it during lift off and landing phases. The location of this platform should be properly evaluated in order to consider safety constraints mainly related to the storage of the propellant used for feeding the capsule propulsion system and to the ground clearance required during the lifting manoeuvres. Considering these characteristics, an ad-hoc spaceport could be envisaged.
Helicopter: The helicopter is considered to be the carrier enabling the capsule to start its mission from a certain altitude, reducing the spacecraft mass (thanks to the propellant mass savings), avoiding the exploitation of expendable multi-stage rockets. The idea of exploiting an existing helicopter is strictly related to the size of the capsule. Moreover, the releasing strategy and the flight procedures should be properly addressed.
Capsule: The capsule can be considered to be the second stage of this complex transportation system. Depending on the releasing altitude, the capsule shall be appropriately propelled in order to reach the target altitude. In order to avoid adding additional complexity, the capsule shall be propelled by one or more rockets able to guarantee the required thrust. The rocket ignition shall be envisaged some TBD seconds after the separation avoiding not to endanger the separation phase. After the rocket burn out, the capsule shall reach the target altitude following a parabolic profile. Then, the capsule shall performed an un-powered but controlled re-entry. This means that the primary propulsion system will not be exploited after the burn-out but a set of parachutes and cold gas thrusters will decelerate the capsule and control its attitude until the approaching phase. Exploiting a properly designed GNC (Guidance and Navigation Control) and ADCS (Attitude Determination Control System) systems, the capsule shall be able to perform a soft vertical landing on the same sea-based platform from which it takes off.</td></tr>
</table>

Table 18: Mission Concept Alternative #2

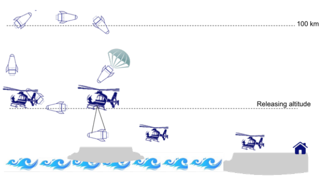
<table>
<tr><td rowspan="4">Helicopter-lifted spaceplane</td><th>Mission scenario</th><th>Mission description</th><th>Comments</th></tr>
<tr><td>
</td><td>A helicopter coming from a ground based infrastructure, reaches a sea-based platform and lift the spaceplane up to a to be defined (TBD) altitude. Then, after the release, the spaceplane is properly propelled to reach the target altitude of 100 km allowing flight participants to experience microgravity for few seconds. During the atmospheric re-entry, a series of parachutes should be deployed to slow down gradually. Additional devices like retro-rockets and control surfaces should be hypothesized to land in the same place from which it takes off</td><td>Like the previous mission concept, this is also characterized by a high level of complexity and risk mainly due to the presence of the helicopter. The spaceplane-like architecture of the spacecraft can allow a higher flexibility in the operations and can host passengers with a higher level of comfort. Both these solutions guarantee a proper view of the Earth if properly designed and equipped. Please notice that additional on ground facilities and rescue subsystems should be envisaged.</td></tr>
<tr><th colspan="3">Mission Elements Description</th></tr>
<tr><td colspan="3">Ground-based platform: The ground-based infrastructures shall support the operations of a heavy–lift helicopter. In order to minimize the turn around time and for economical reasons, the platform shall be located in a coastal region, in the proximity of a sea-based platform from which the core of the mission is planned to start.
Sea-based platform: The sea-based platform shall host the infrastructures to accommodate and maintain the spaceplane and support it during lift off and landing phases. The location of this platform should be properly evaluated in order to consider safety constraints mainly related to the storage of the propellant used for feeding the spaceplane propulsion system and to the ground clearance required during the lifting manoeuvres. Considering these characteristics, an ad-hoc spaceport could be envisaged.
Helicopter: The helicopter is consider to be the carrier enabling the spaceplane to start its mission from a certain altitude, reducing the spacecraft mass (thanks to the propellant mass savings), avoiding the exploitation of expendable multi-stage rockets. The idea of exploiting an existing helicopter is strictly related with the sizing of the capsule. Moreover, the releasing strategy and the flight procedures should be properly addressed.
Spaceplane: The spaceplane can be considered to be a second stage of this complex transportation system. Depending on the releasing altitude, it shall be properly propelled in order to reach the target altitude. The possibility of ignite the propulsion system at a certain altitude is a non-negligible advantage in terms of mass savings complexity of the spaceplane. Indeed, the possibility of exploiting a lifter-helicopter will avoid implementing a demanding and problematic propulsion subsystem aimed at performing a vertical take off from the platform. Considering this mission scenario, a single stage with a single propulsion system seems to be the most promising solution. In particular, advanced air-breathing propulsion will guarantee the minimum fuel mass but considering the level of maturity of the technology and the short duration of the mission (please, consider that the advantages of airbreathing with respect to rockets increase with the mission duration), a rocket engine shall be considered. It is clear that this choice impacts not only the vehicle itself but also the sea-based platform facility for the storage of the propellant and also the operational procedures and platform location. Considering the envisaged scenario, the spaceplane should be properly equipped with a landing gear able to support a vertical landing, after the re-entry and descent phases.</td></tr>
</table>

Table 19: Mission Concept Alternative #3

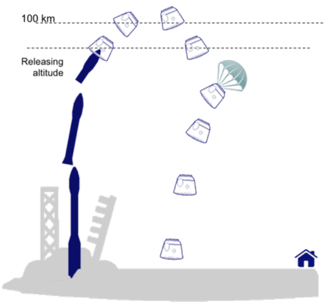
<table>
<tr><td rowspan="4">Rocket launched capsule</td><th>Mission scenario</th><th>Mission description</th><th>Comments</th></tr>
<tr><td>
</td><td>A capsule is launched using an existing launcher. Exploiting the thrust provided by each of the different stages, the capsule should be able to reach the 100 km of altitude after the release. During the atmospheric re-entry, a series of parachutes should be deployed to slow down gradually. Additional devices like retro-rockets and control surfaces should be hypothesized to land in the same place from which it takes off.</td><td>The exploitation of a traditional multi-stages rocket to bring a capsule up to a certain altitude will reduce the problems related to on ground infrastructure. However, this solution will not fit with the need for guaranteeing a routine service, because of several reasons (e.g. expendable stages, the need of launch windows, very high cost per mission, etc…). Conversely, this solution appears very promising from the spacecraft design point of view. Indeed, a valuable reduction of power and mass budget can be obtained. Passengers requirements on comfort could be not at all satisfied, especially during the descending parabolic phase.</td></tr>
<tr><th colspan="3">Mission Elements Description</th></tr>
<tr><td colspan="3">Ground infrastructure: The ground-based infrastructures shall support the operations of a launcher. This means that existing space centres shall be selected or new ad-hoc facilities shall be built. The facility shall accommodate the required amount of fuel and shall provide workshops for maintenance. The problem of guaranteeing “routine” service shall be addressed.
Launcher: Depending on the sizing of the capsule, the launcher could be an existing or under-development one, or an enhanced version of an existing one shall be proposed. The use of a multi-stages rocket dramatically simplifies the architecture of the capsule and in particular of its propulsive system.
Capsule: With the possibility of exploiting the different stages of an expendable rocket, the capsule can be un-powered, having only thrusters to guarantee manoeuvrability, especially during re-entry and descent phases. Precise landing is also required and this implies the need for implementing a deceleration subsystem (parachute and thrusters) and GNC and ADCS subsystems. The system could be very simple and existing capsules could be taken as reference. For simplicity, the landing gear could be substituted with inflatable bags but the bouncing on ground could be non acceptable for non trained people</td></tr>
</table>

Table 20: Mission Concept Alternative #4

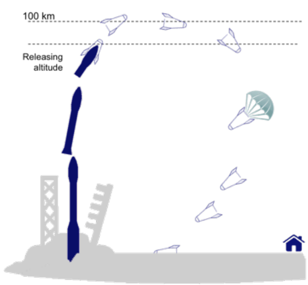
<table>
<tr><td rowspan="4">Rocket launched spaceplane</td><th>Mission scenario</th><th>Mission description</th><th>Comments</th></tr>
<tr><td></td><td>A spacecraft is launched using an existing launcher. Exploiting the thrust provided by each of the different stages, the capsule should be able to reach the 100 km of altitude after the release. During the atmospheric re-entry, a series of parachutes should be deployed to slow down gradually. Additional devices like retro-rockets and control surfaces should be hypothesized to land in the same place from which it takes off.</td><td>The exploitation of a traditional multi-stages rocket to bring a capsule up to a certain altitude will reduce the problems related to on ground infrastructure. However, this solution will not fit with the need for guaranteeing a routine service, because of several reasons (e.g. expendable stages, the need of launch windows, very high cost per mission, etc…). Conversely, this solution appears very promising from the spacecraft design point of view. Indeed, a valuable reduction of power and mass budget can be obtained. After the release, the spaceplane can safely accommodate passengers also during the re-entry phase.</td></tr>
<tr><th colspan="3">Mission Elements Description</th></tr>
<tr><td colspan="3">Ground infrastructure: The ground-based infrastructures shall support the operations of launcher. This means that existing space centres shall be selected or new ad-hoc facilities shall be built.
Launcher: Depending on the sizing of the capsule, the launcher could be an existing or under-development one, or an enhanced version of an existing one shall be proposed. The use of a multi-stages rocket dramatically simplifies the architecture of the capsule and in particular of its propulsive system.
Spaceplane: The spaceplane shall be designed in order to fit into the launcher upper stage. With the possibility of exploiting the different stages of an expendable rocket, the spaceplane design can be simplified. It can be un-powered, having only thrusters to guarantee manoeuvrability, especially during re-entry and descent phases. Precise landing is also required and this implies the need for implementing a deceleration subsystem (parachute and thrusters) and GNC and ADCS subsystems.</td></tr>
</table>

Table 21: Mission Concept Alternative #5

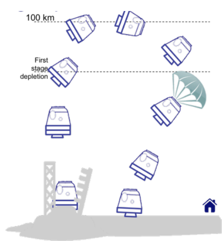
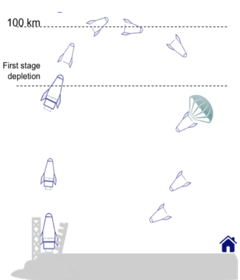
<table>
<tr><td rowspan="4">Powered capsule</td><th>Mission scenario</th><th>Mission description</th><th>Comments</th></tr>
<tr><td>
</td><td>A capsule (single or multi-stages) is able to vertically take-off exploiting the thrust produced by its embedded propulsion system. In order to accomplish the mission requirements, the capsule should be able to reach the 100 km of altitude after the release. During the atmospheric re-entry, a series of parachutes should be deployed to slow down gradually. Additional devices like retro-rockets and control surfaces should be hypothesized to land in the same place from which it takes off.</td><td>This mission concept deletes all the problems related to physical and operational interfaces with the launcher but a very high level of complexity of the spacecraft subsystems characterizes it. Furthermore, this mission concept option has a moderate impact on the infrastructures and can enhance the frequency of the service provided. The level of comfort for the passengers is not very high.</td></tr>
<tr><th colspan="3">Mission Elements Description</th></tr>
<tr><td colspan="3">Ground infrastructure: The ground-based infrastructures shall support the operations of a Single Stage or Two Stage capsule-like system. Depending on the capsule architecture, the Ground Infrastructure could vary from a complex centre, similar to a space one but with ad-hoc launch facility, to a simple prepared pad from which the capsule can autonomously lift-off, exploiting its own landing gear legs. The facility shall accommodate the required amount of fuel and shall provide workshops for maintenance. The problem of guaranteeing “routine” service shall be addressed.
Capsule: This scenario allows different architectures for the capsule system architecture. Indeed, depending on the staging strategy and on the degree of reusability of the overall transportation system, different alternatives for the propulsion system could be envisaged. After trade off analyses, the alternative envisaging a single stage capsule, completely reusable and rocket propelled has been preferred. The use of a rocket-based propulsion system since the beginning of the mission implies the construction of ad-hoc on ground facilities and a widening of the clearance area required for the operations.</td></tr>
</table>

Table 22: Mission Concept Alternative #6 (selected)

<table>
<tr><td rowspan="4">Powered Spaceplane</td><th>Mission scenario</th><th>Mission description</th><th>Comments</th></tr>
<tr><td>
</td><td>A spaceplane (single or multi-stages) is able to vertically take-off exploiting the thrust produced by its embedded propulsion system. In order to accomplish the mission requirements, the vehicle should be able to reach the 100 km of altitude after the release. During the atmospheric re-entry, a series of parachutes should be deployed to slow down gradually. Additional devices like retro-rockets and control surfaces should be hypothesized to land in the same place from which it takes off.</td><td>Similarly to the previous mission concept, also in this case, interface criticalities with the launcher do not exist. Furthermore, this mission concept option has a moderate impact on the infrastructures and can enhance the frequency of the service provided. The spaceplane system configuration can be very complex and the related mass and power budget can be very demanding. Please consider that this solution is the best option from the point of view of the passengers, because very similar to a typical aircraft aimed at transportation purposes.</td></tr>
<tr><th colspan="3">Mission Elements Description</th></tr>
<tr><td colspan="3">Ground infrastructure: The ground-based infrastructures shall support the operations of a Single Stage system able to automatically performed take off and landing manoeuvres. The spaceport could be a simple prepared pad from which the system can autonomously lift-off, exploiting its own landing gear legs. The main problems could be related to the storage of propellant into the facility and the logistic and maintenance support.
Spaceplane: The envisaged spaceplane shall be a Single Stage, which shall be able to perform a vertical take off in tail-sitting or (more preferable) in Harrier-like position. In order to overcome exiting environmental regulations forbidding the use of rocket propulsion under a certain altitude, an airbreathing propulsion system will be exploited during the take off and landing manoeuvres. The airbreathing propulsion system will be exploited up to its ceiling altitude, when rocket will be ignited. Than, the spaceplane will be powered by a rocket motors to reach the target altitude. After the parabolic phase and a first part of un-powered re-entry, the airbreathing propulsion system could be re-started in order to enhance the accuracy of the descent and landing phases. Controlled and precise landing manoeuvres could be carried out.</td></tr>
</table>

3.7 System Level Analysis (Case Study).

Following the same approach previously described and then applied up to the selection of the most promising mission alternative, the selected mission concept can be detailed and a new list requirements elicited. In particular, in this section, it is possible to notice also the process that allows to revise the functional requirements, making possible updates. In addition, as it is possible to see in Figure 75 and Figure 76, the exploitation of a software based requirements management tool allow the engineers to trace the different versions of each requirement. This is a fundamental feature, that would be especially exploited in case of numerical requirements that may change the associate numerical value all along the design process and even more, during the overall product life cycle.

The Section starts reporting the complete Functional Tree (developed up to system level) and allowing the elicitation of the first list of segment and system level functional requirements. Then, after a proper allocation to system level components through connection matrixes, the Product Tree can be updated with a new level and the first list of requirements can be easily updated.

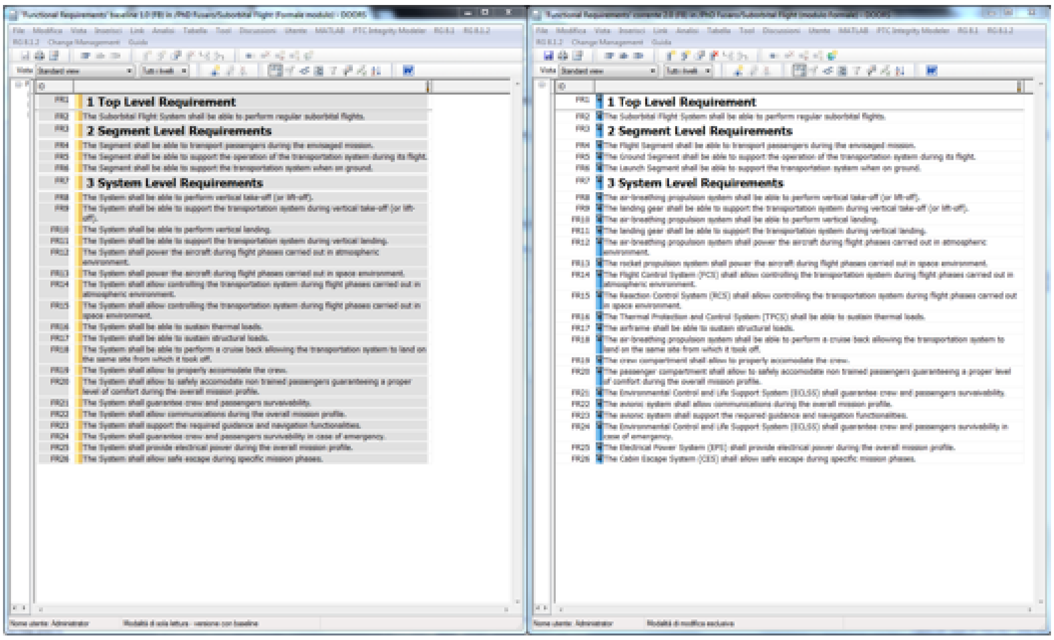


Figure 75: Requirements Baselines comparison

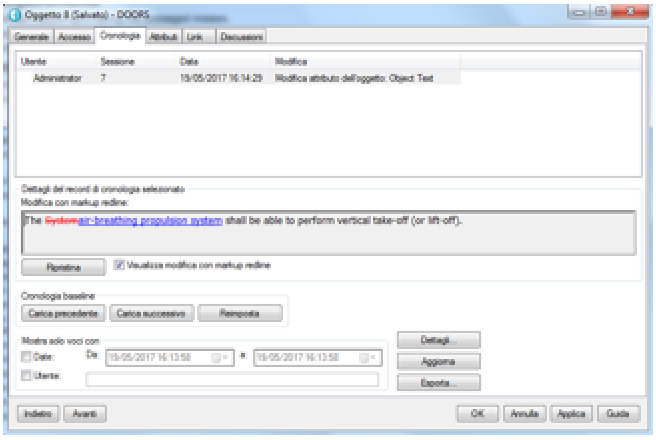


Figure 76: Requirements track changes

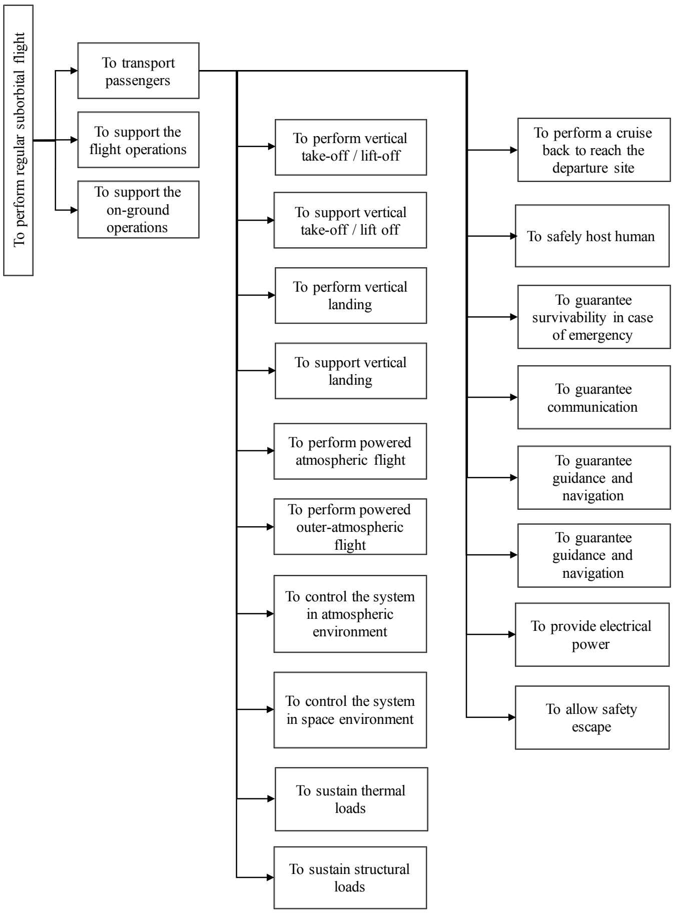


Figure 77: Functional Tree (System Level)

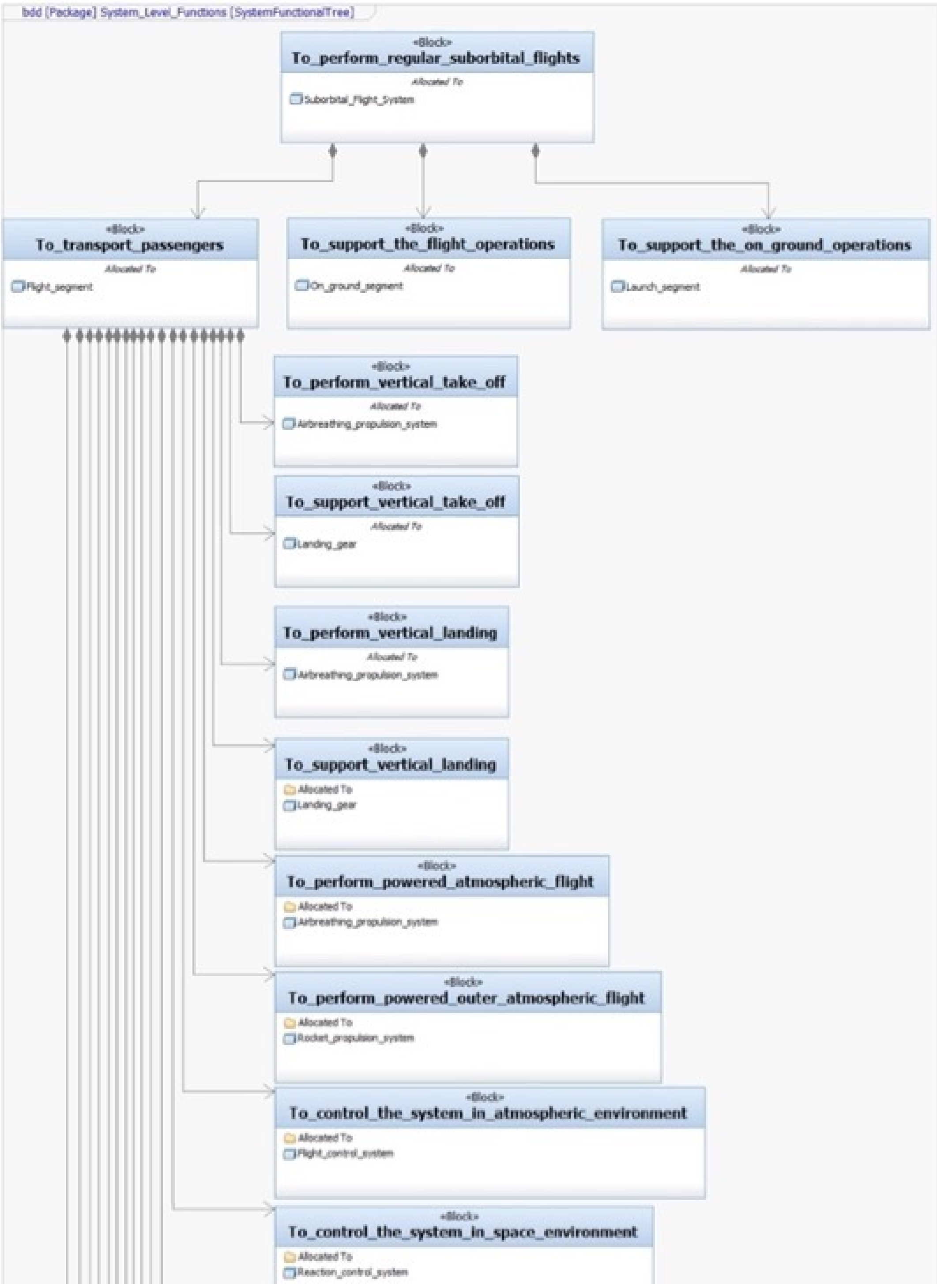


Figure 78: Functional Tree (System Level). Example of implementation with BDD

Functional Requirements (Segment Level)

FR 1. The Segment shall be able to transport passengers during the envisaged mission
FR 2. The Segment shall be able to support the operation of the transportation system during its flight
FR 3. The Segment shall be able to support the transportation system when on-ground

Functional Requirements (System Level)

FR. 1.1. The system shall be able to perform vertical take-off (or lift-off)
FR. 1.2. The system shall be able to support the transportation system during vertical take-off (or lift-off).
FR. 1.3. The system shall be able to perform vertical landing
FR. 1.4. The system shall be able to support the transportation system during vertical landing
FR. 1.5. The system shall power the aircraft during flight phases carried out in atmospheric environment
FR. 1.6. The system shall power the aircraft during flight phases carried out in space environment
FR. 1.7. The system shall allow to control the transportation system during flight phases carried out in atmospheric environment
FR. 1.8. The system shall allow to control the transportation system during flight phases carried out in space environment
FR. 1.9. The system shall be able to sustain thermal loads
FR. 1.10. The system shall be able to sustain structural loads
FR. 1.11. The system shall be able to perform a cruise back allowing the transportation system to land on the same site from which it took off
FR. 1.12. The system shall allow to properly accommodate the crew
FR. 1.13. The system shall allow to safely accommodate non trained passengers guaranteeing a proper level of comfort during the overall mission profile.
FR. 1.14. The system shall guarantee crew and passengers survivability
FR. 1.15. The system shall allow communications during the overall mission profile
FR. 1.16. The system shall support guidance and navigation required functionalities

FR. 1.17. The system shall guarantee crew and passengers survivability in case of emergency

FR. 1.18. The system shall provide electrical power during the overall mission profile

FR. 1.19. The system shall allow safety escape during specific mission phases

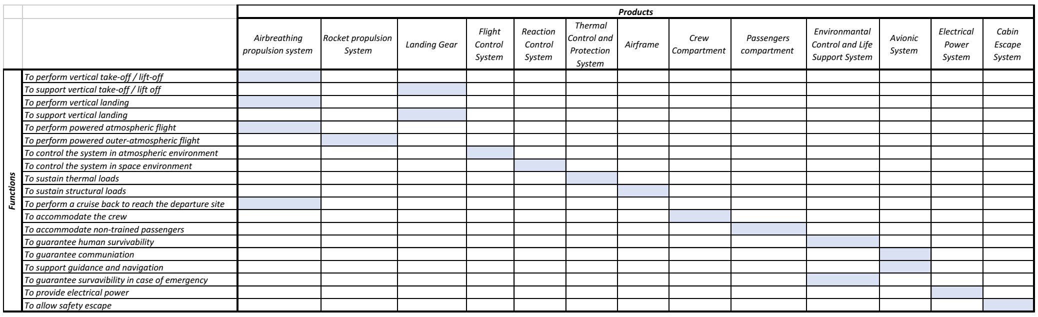


Figure 79: Connection Matrix at System Level

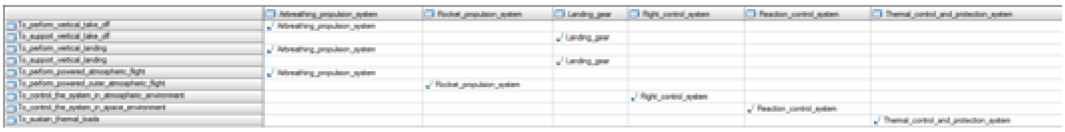


Figure 80: Connection Matrix at System Level - MBSE

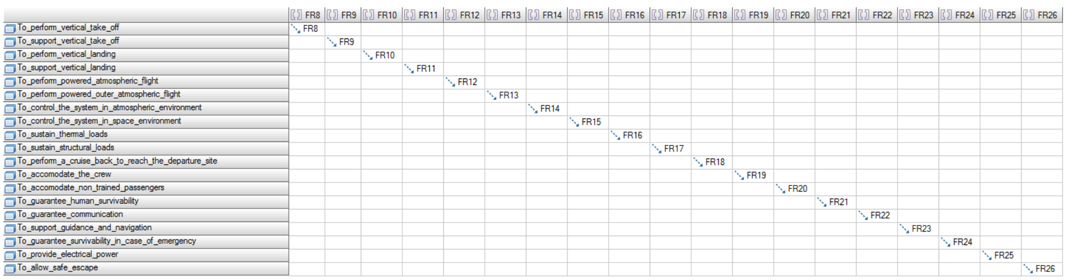


Figure 81: Connection Matrix in MBSE (Functions and Functional Requirements) at System Level

Functional Requirements (Segment Level)- Refined

FR 4. The flight segment shall be able to transport passengers during the envisaged mission

FR 5. The ground segment shall be able to support the operation of the transportation system during its flight

FR 6. The launch segment shall be able to support the transportation system when on-ground

Functional Requirements (System Level) - Refined

FR. 1.1. The air-breathing propulsion system shall be able to perform vertical take-off (or lift-off)

FR. 1.2. The landing gear shall be able to support the transportation system during vertical take-off (or lift-off).

FR. 1.3. The air-breathing propulsion system shall be able to perform vertical landing

FR. 1.4. The landing gear shall be able to support the transportation system during vertical landing

FR. 1.5. The air-breathing propulsion system shall power the aircraft during flight phases carried out in atmospheric environment

FR. 1.6. The rocket propulsion system shall power the aircraft during flight phases carried out in space environment

FR. 1.7. The Flight Control System (FCS) system shall allow to control the transportation system during flight phases carried out in atmospheric environment

FR. 1.8. The Reaction Control System (RCS) shall allow to control the transportation system during flight phases carried out in space environment

FR. 1.9. The Thermal Protection and Control System (TPS/TCS) shall be able to sustain thermal loads

FR. 1.10. The airframe shall be able to sustain structural loads

FR. 1.11. The air-breathing propulsion system shall be able to perform a cruise back allowing the transportation system to land on the same site from which it took off

FR. 1.12. The crew compartment shall allow to properly accommodate the crew

FR. 1.13. The passengers compartment shall allow to safely accommodate non trained passengers guaranteeing a proper level of comfort during the overall mission profile

FR. 1.14. The Environmental Control and Life Support System (ECLSS) shall guarantee crew and passengers survivability

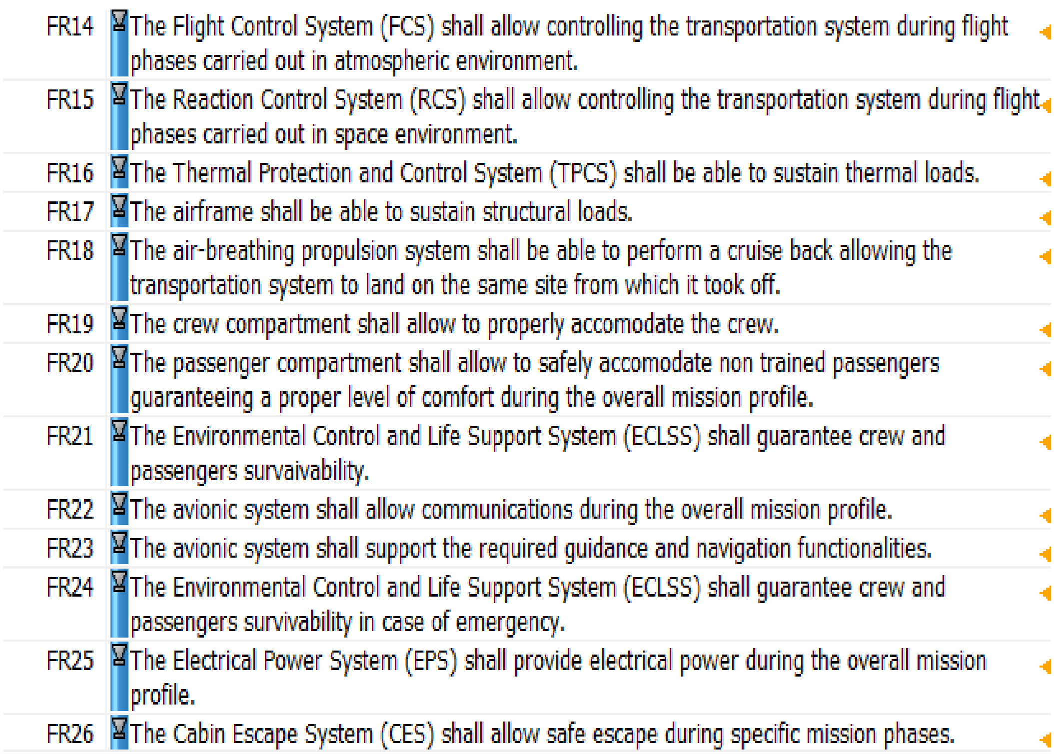
FR. 1.15. The avionic system shall allow communications during the overall mission profile

FR. 1.16. The avionic system shall support guidance and navigation required functionalities

FR. 1.17. The Environmental Control and Life Support System (ECLSS) shall guarantee crew and passengers survivability in case of emergency

FR. 1.18. The Electrical Power System (EPS) shall provide electrical power during the overall mission profile

FR. 1.19. The Cabin Escape System (CES) system shall allow safety escape during specific mission phases



FR14	The Flight Control System (FCS) shall allow controlling the transportation system during flight phases carried out in atmospheric environment.
FR15	The Reaction Control System (RCS) shall allow controlling the transportation system during flight phases carried out in space environment.
FR16	The Thermal Protection and Control System (TPCS) shall be able to sustain thermal loads.
FR17	The airframe shall be able to sustain structural loads.
FR18	The air-breathing propulsion system shall be able to perform a cruise back allowing the transportation system to land on the same site from which it took off.
FR19	The crew compartment shall allow to properly accomodate the crew.
FR20	The passenger compartment shall allow to safely accomodate non trained passengers guaranteeing a proper level of comfort during the overall mission profile.
FR21	The Environmental Control and Life Support System (ECLSS) shall guarantee crew and passengers survaivability.
FR22	The avionic system shall allow communications during the overall mission profile.
FR23	The avionic system shall support the required guidance and navigation functionalities.
FR24	The Environmental Control and Life Support System (ECLSS) shall guarantee crew and passengers survivability in case of emergency.
FR25	The Electrical Power System (EPS) shall provide electrical power during the overall mission profile.
FR26	The Cabin Escape System (CES) shall allow safe escape during specific mission phases.

Figure 82: Example of Functional Requirements implementation in DOORS®

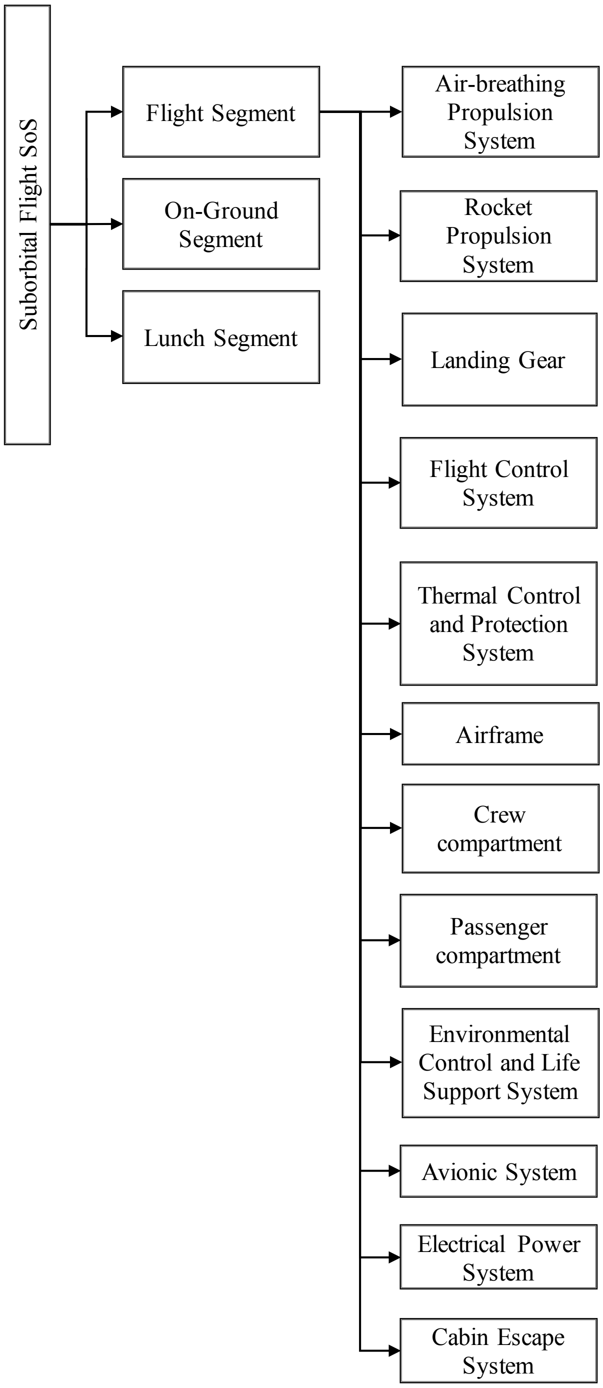


Figure 83: Product Tree (System Level)

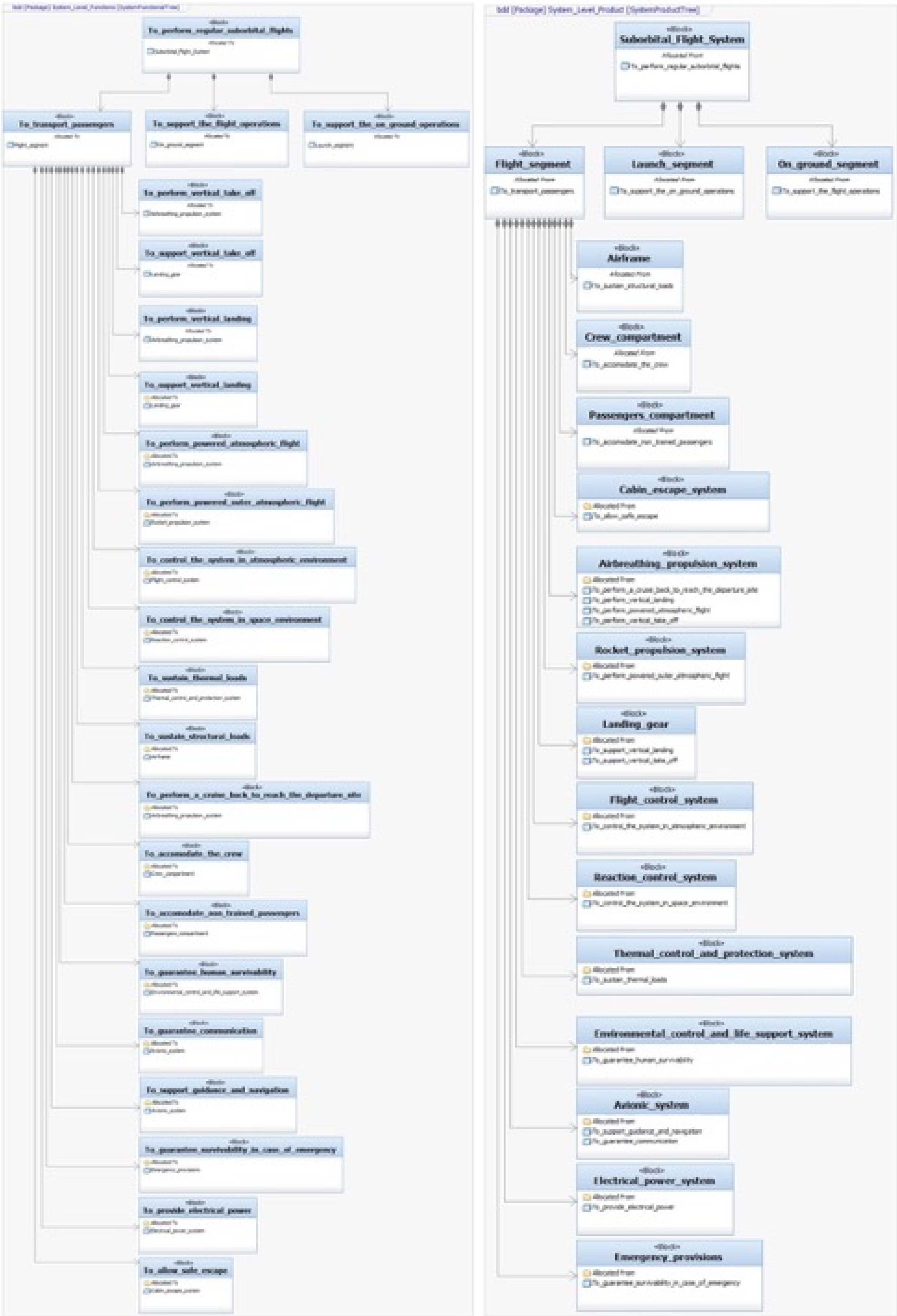


Figure 84: Product Tree (System Level) – with BDD

The trade-off analysis to properly select the optimal elements to compose the SoS architecture cannot be carried out without considering the way in which these elements could be operated together within the overall SoS. At this purpose, the Concept of Operations should also be sketched, at least highlighting the major mission phases and a feasible communication architecture.

Table 23: Mission Phases

Mission Phase	*Starting altitude*	*Ending altitude*
Take off	0 m	20 m
1st climb segment (transition)	20 m	300 m
2nd climb segment	300	18000
3rd climb segment	18000 m	60000 m
4th climb segment	60000 m	100000 m
Re-entry (Ballistic)	100000m	75000 m
Powered Re-entry	75000 m	15000 m
Cruise	15000 m	15000 m
Descent	15000 m	20 m
Landing	20 m	0 m

Considering these tables, it is worth to notice that the high level operational modes here described, refer to the spacecraft system. Indeed, it is not possible to define the operational modes without having identified the system to which they refer to. Moreover, the identified operational modes are:

- Un-powered: this is the operational mode during which the spacecraft does not use neither the air-breathing engines nor the rocket one and its motion its governed by the inertial forces.
- Powered: this is the operative mode in which the spacecraft exploits its air-breathing engines.
- Rocketed: this is the operative mode in which the rocket engine is ignited.
- Safety: is the typical operational mode that is encountered each time a minor failure or malfunctioning is identified. Depending on the phase in which it happens and the associated level of risk, the trajectory could be modified and the spacecraft subsystems could change their operative modes to overcome the problem.
- Escape: it is the operative mode related to the highest level of risk. In this case, the spacecraft is considered no more able to carry out the nominal mission. The spacecraft is separated into two pieces. The small one, corresponding to the front fuselage, contains the crew and related vital subsystems and should be designed in order to allow the crew and passengers survival, landing, after a ballistic, un-powered phase.

Table 24: Modes of Operation (Vehicle Level)

Mission Phase	***Un-powered***	***Powered***	***Rocketed***	***Safety***	***Escape***
Take off		X		X	
1st climb segment		X		X	X
2nd climb segment		X		X	X

3rd climb segment			X	X	X
4th climb segment	X			X	X
Re-entry (Ballistic)	X			X	X
Powered Re-entry		X		X	X
Cruise		X		X	X
Descent		X		X	X
Landing		X		X	

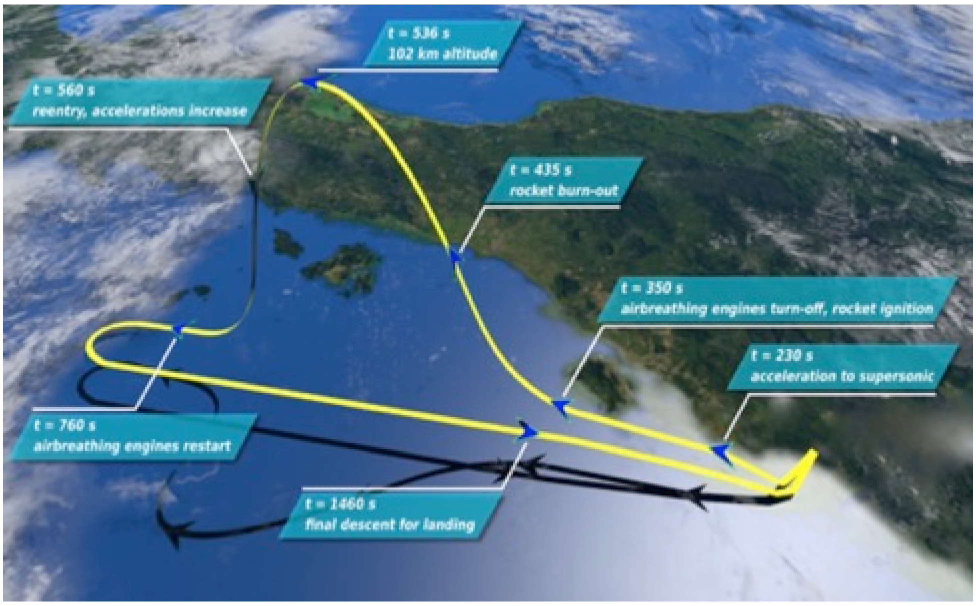


Figure 85: Pictorial View of a possible Mission Profile (Fusaro, 2015)

References

Akao, J. et al (2003) The leading edge in QFD: past, present and future. *International Journal of Quality & Reliability Management* 20.1 (2003): 20-35.

Chan, L., et al. (2003). Quality function deployment: A literature review. *European Journal of Operational Research* 143.3 (2002): 463-497.

Civil Aviation Authority (2018), Summary and Government response to the consultation on criteria to determine the location of a UK spaceport; Department for Transport, *Department for business Innovation and skills*.

Cresto Aleina, S., et al. (2015) The importance of technology roadmaps for a successful future in space exploration, *9th IAA Symposium on the future of space exploration*. Torino (Italy).

Cresto Aleina, S., et al. (2016) Effective methodology to derive strategic decisions from ESA exploration technology roadmaps. *Acta Astronautica*, 126, pp. 316-324.

Cresto Aleina, S., et al. (2017) Technology roadmaps derivation methodology for European hypersonic and re-entry space transportation systems. *21st AIAA International Space Planes and Hypersonics Technologies Conference, Hypersonics 2017*, 16 p.

Fusaro, R., et al (2015). Conceptual design and operations of a crewed reusable space transportation system. *Proceedings of the International Astronautical Congress,* IAC Jerusalem (Israel), 11, pp. 8895-8909.

Fusaro, R., et al (2017). Conceptual design of a crewed reusable space transportation system aimed at parabolic flights: stakeholder analysis, mission concept selection, and spacecraft architecture definition. *CEAS Space Journal*, 9 (1), pp. 5-34. DOI: 10.1007/s12567-016-0131-7

Fusaro, R., et al. (2017b) Database on hypersonic transportation systems: A versatile support for the technology roadmap generation and conceptual design activities. *21st AIAA International Space Planes and Hypersonics Technologies Conference, Hypersonics 2017*, 16 p.

Jakhu, R. S. et al. (2011). The Need for an Integrated Regulatory Regime for Aviation and Space: ICAO for Space?. *Vol. 7. Springer Science & Business Media*, 2011.

NASA (2007) Systems Engineering Handbook, *NASA/SP-2007-6105 Rev1, National Aeronautics and Space Adminstration*, NASA Headquarters Washington, D.C. 20546.

Santoro, F., et al. (2015). Approaches to development of commercial spaceport and associated ground segment driven by specific spaceplane vehicle and mission operatio requirements. Proceedings of the International Astronautical Congress, Jerusalem (Israel) IAC, 13, pp. 9908-9922.

Santoro, F., et al. (2016), Advances in Italian spaceports identification as infrastructures for suborbital flight activities and mission profiles (2016) *Proceedings of the International Astronautical Congress, IAC*, Guadalajara (Mexico).

Tauri Group (2012). A 10-Year Forecast of Market Demand. The Tauri Group under contract with the Federal Aviation Administration and Space Florida.

Viscio, M.A, et al. (2013). A methodology to support strategic decisions in future human space exploration: From scenario definition to building blocks assessment. *Acta Astronautica*, 91, pp. 198-217

Viscio, M.A, et al. (2014). A methodology for innovative technologies roadmaps assessment to support strategic decisions for future space exploration. *Acta Astronautica*, 94 (2), pp. 813-833.

Chapter 4

Hypersonic System Design

4.1 Possible vehicle architectures

4.1.1 Classification criteria

Considering all the past and currently under-development projects, it is very difficult to find a unique parameter for the classification of vehicles dealing with hypersonic. Indeed, depending on the specific discipline, they can be grouped following different criteria. The easiest categorizations are based on the operative environment (Fusaro, 2017) or on the maximum achievable Mach number. However, an interesting classification criterion has been proposed by Hirschel in several of his works (Hirschel, 2005) and (Hirschel, 2009) and also used by other authors (Weiland, 2014) and (Kuczera, 2011). This hybrid categorization mixing together configurational characteristics, propulsive system and mission profiles. In order to include suborbital vehicles within this classification, the following categorization is adopted:

- Re-entry Vehicles (RV)
    - Winged re-entry vehicles (W-RV)
    - Non winged re-entry vehicles (NW-RV)
- Ascent and re-entry vehicles (ARV)
    - Orbital ascent and re-entry vehicles (O-ARV)
    - Suborbital ascent and re-entry vehicles (SO-ARV)

- Cruise and acceleration vehicles (CAV)

Table 25 shows some example of past and currently under-development projects for each of these families.

Table 25: Vehicle aerothermodynamic categorization (examples) (Hirschel, 2005)

	W-RV	NW-RV	O-ARV	SO-ARV	CAV
Reference Vehicle	X-33,34,37,38 HERMES HOPE PHOENIX PRORA	EXPERT IRDT COLIBRI BENT-BICONE IXV Pre-X ORION	Space Shuttle BURAN NASP HOTOL X-33 FESTIP	SpaceLoft XL STIG LynxMark SpaceShipTwo Xaero New Shepard	SAENGER ELAC LAPCAT SpaceLiner

In this context, considering that the aim of the present section is to provide an overview of the major features of hypersonic vehicles mainly in terms of vehicle architecture and layout, the various initiatives are classified depending on high-level characteristics such as the staging strategy, the take-off strategy and the Lift-over-Drag parameter. Each of these characteristics deeply affects both the vehicle and the missions, with noticeable consequences at SoS level. These characteristics are detailed in the following subsections.

4.1.2 Staging strategy

The number of stages of a transportation system is a macroscopic element of the layout that can be easily recognized at a first look of the overall system. Conversely, the staging strategy is more complex to be understood, requiring an integrated consideration of the vehicle's stages, their main subsystems and the mission profile. Indeed, the staging strategy for a so complex aerospace system shall take into account not only the number of stages which the entire transportation system consists of, but it is also affected by the way in which the propulsive system and the propellant feeding strategy are exploited. The different missions of each single stage shall be taken into account too.

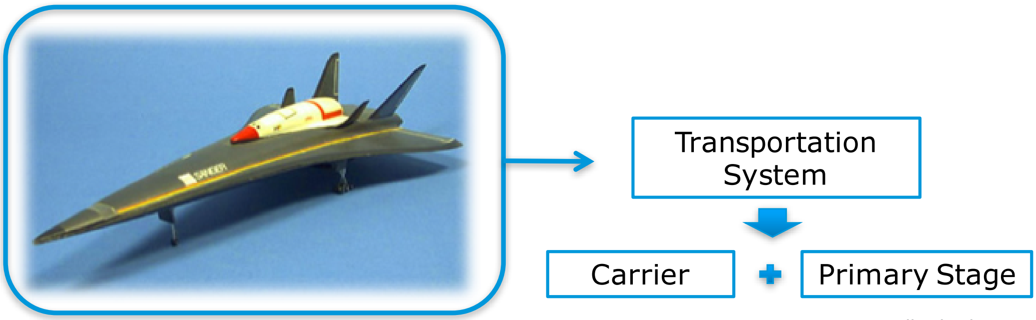


Figure 86: Multiple stages transportation system

4.1.2.1 Staging strategy alternatives

As far as the number of stages is concerned, looking at all the different configurations emerged from the historical review of past and on-going initiatives dealing with hypersonic, the following alternatives can be identified:

- **Single stage**. The Transportation System is composed of a single vehicle that should contain all the subsystems enabling all the capabilities required to carry out future the mission. In particular, the case of a fully reusable single stage vehicle could be considered the “ideal” configuration. Indeed, it would be very similar to a conventional aircraft, especially from the point of view of the on-ground operations and logistic, avoiding the technical additional complexities related to the integration of more stages and diminishing the risk connected to the separation phase. Conversely, the major drawback of such configuration is related to the higher take-off gross weight. Rocketplane is currently trying to overcome this problem with its concept vehicle Pioneer. It aims at diminish the fuel mass stored on-board, suggesting an air-refuelling. In this way the maximum take-off gross weight can be drastically reduced as well as the risk of incidents at take-off related to the on-board presence of dangerous propellant. Of course, a proper Technology Readiness Level (TRL) should be reached in the field of air-refuelling of liquid hydrogen propellant. This aspect can also have a deep impact of the spaceport design and location selection as it would be explained in the following subsection.
- **Two Stages.** It is considered the best compromises between weight reduction and increase in complexity. In case of a two stages, it is possible to refer to the

overall integrated vehicle as the Transportation System. It consists of a first stage referred to as carrier and a second stage that is the vehicle that really perform the mission operations, for this reason usually referred to as the primary stage. Among all the past and currently under-development initiatives, there are concepts in which the carrier vehicle is a commercial operative aircraft (civil or military). This is a commercial choice aimed at minimizing costs, devoting all the economic and technical efforts at the design and development of the primary stage. This could also be a good solution especially in case of demo or test missions.

In a Two Stages transportation system, different propulsive strategies could be envisaged and Table 26 summarizes possible alternatives. Some of the configurations, resulting from direct composition of staging strategy and propulsive strategy, are clearly unfeasible from the technical point of view. In particular, two of them don't seem to be a reasonable alternative. The first is the Conf. 3.1, consisting of a configuration hosting the propulsion system in the second stage only with the possibility of using it only when attached to the first stage containing the necessary propellant. The only case of application could be the one in which the first stage is a sort of expendable tank and the capability to host the engine in the second stage can allow a great saving in terms of costs (construction and operations). The second not very practical configuration is Conf. 1.2 that proposes a first stage with propulsive element only and a complete second stage. This configuration requires the second stage to host all the amount of propellant required to feed both stages, with undesirable increase in the second stage mass.

A part from these configurations, differently from the first stage that could really have different design alternatives, the second stage could be either an autonomous vehicle (with engines and tanks) or a vehicle without any propulsive capabilities performing an unpowered re-entry (Conf. 2.4). In this case, the first stage should obviously be autonomous providing all the capabilities to allow the vehicle to reach the desired target altitude. This is exactly the case of IXV mission. A similar case is the one (Conf. 3.2) in which the powered second stage is associated to a first one that acts as tank. In this case, the optimal strategy should be the one in which before the separation, a propellant transfer should guarantee a re-filling of the tank of the second stage. Another interesting alternative is the one in which both the stages have propulsive capabilities (Conf. 2.2). In this case, in order to save costs and allows services on regular bases, the most convenient case is the one of a fully reusable transportation system. In this case the second stage is really optimized for its peculiar mission. Conf. 3.1 is the one in which the first stage acts as an over-boost for the very first phases of the missions. Please note that in this case, the tank of the second stage should be sized considering the overall mission profile and not only the trajectory legs following the separation. This has the only advantage of a simplified configuration, as far as the first stage is concerned, limiting the impact in terms of mass related to the additional engines required to fulfil the take off and climb requirements.

- **Three stages.** This group has only few examples of conceptual design activities mainly carried out in the Soviet Union. The increased number of stages implies higher level of complexity and costs but can allow to increase the maximum altitude and payload capabilities, desirable aspects for the missions devoted to enhancing the access to space possibilities but difficult to be implemented in suborbital or a P2P mission.

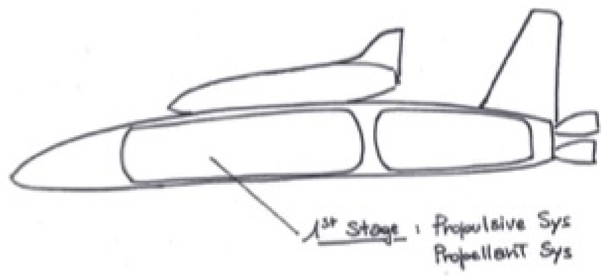


Figure 87: Staging Strategy Alternative – Conf. 2.4 (b)

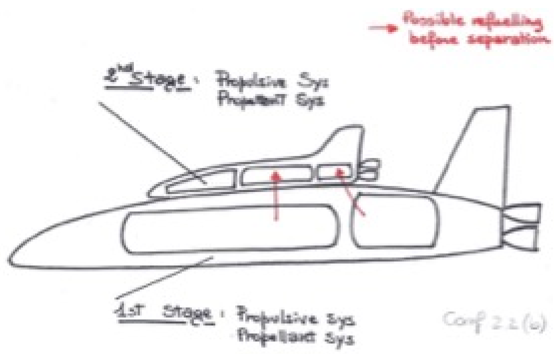


Figure 88: Staging Strategy Alternative – Conf. 2.2 (b)

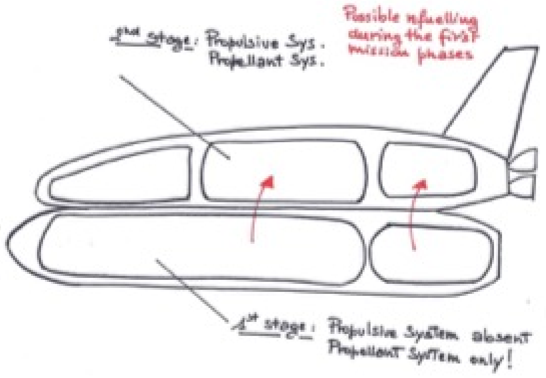


Figure 89: Staging Strategy Alternative – Conf. 3.2

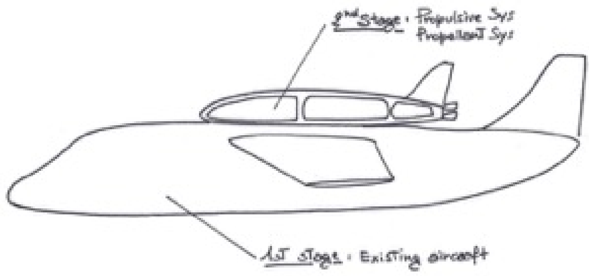


Figure 90: Staging Strategy Alternative – Conf. 2.2 (a)

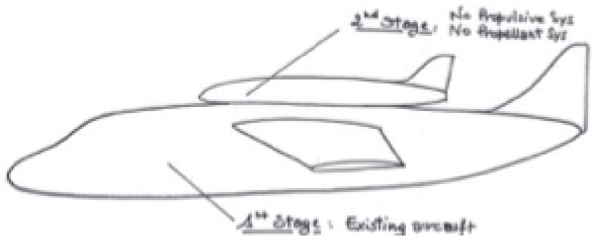


Figure 91: Staging Strategy Alternative – Conf. 2.4 (a)

Table 26: Propulsive alternatives for a two stages transportation system

Second Stage \ First Stage	Propulsive System only	Propulsive Sys and Propellant Sys(existing carrier)	Propulsive Sys and Propellant Sys (To be developed)	Propellant System only
Propulsive System only	Unfeasible	Unfeasible	Unfeasible	Conf. 3.1
Propulsive Sys and Propellant Sys	Conf. 1.2	Conf. 2.2 (a)	Conf. 2.2 (b)	Conf. 3.2
Propellant System only	Unfeasible	Unfeasible	Unfeasible	Unfeasible
No Propulsive and No Propellant systems	Unfeasible	Conf. 2.4 (a)	Conf. 2.4 (b)	Unfeasible

It has to be noticed that each stage, depending on the high level requirements, could be reusable, semi-reusable or expendable. Moreover, this classification can be sufficient in order to host supersonic vehicles launched as payload of existing launch vehicles (Vega, Arianne, etc…). Indeed, the launch vehicle, even if it consists itself of several stages, can be considered as the carrier of a two stages configuration.

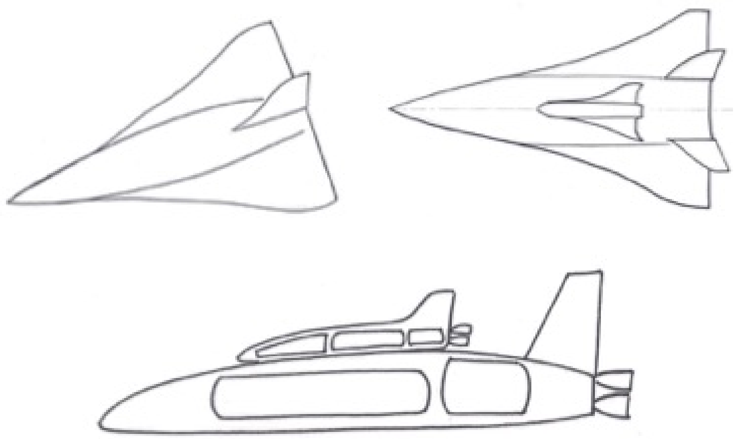


Figure 92: Pictorial representation of Staging Strategy Alternatives

4.1.2.2. Staging strategy trade-off

In order to understand the best alternative in terms of staging strategies, a trade-off analysis can be carried out. To this purpose, it is important to identify the major elements that impact on the selection of the optimal staging strategy. In this case, complexity, costs and safety have been selected as Figure of Merits (FoMs).

Unfortunately, due to the high level of these analyses, it is very difficult to evaluate all them mathematically but it is necessary to avoid excessive subjectivity in the decisional process. A preparatory step to the formulation of equations that can allow the evaluation of these FoMs at so high design level has been carried out, analysing the most impacting design parameters, and it is reported in Table 27.

Table 27: Impact of design parameters on FoMs for the staging strategy trade-off

Figure of Merit	Design Parameters impacting on the FoM evaluation	Comments
Complexity	Number of stages	The number of stages deeply affect the complexity of the transportation system with an impact that is proportional to the number of stages that should be ad-hoc designed and built.
	Presence of propulsive system on each stage	The propulsive subsystem is one of the most complex in a hypersonic transportation system and it is a key factor for the definition of the complexity of the spaceplane. The presence of engines in a stage is also impacting on the maintenance and logistic activities required and on the related turn-around time.
	Presence of propellant tanks on each stage	The presence of propellant requires the construction and integration of tanks in the stage. The impact on the complexity is relevant even if lower wrt the propulsive system.
	Presence of cross-feed between stages	In case of a multi-stages vehicle, the presence of tank in both stages can require the construction of proper cross-feed subsystems. This could be very impacting on the complexity of the overall transportation system.
	Exploitation of existing	The exploitation of existing stages diminishes the complexity of the design and the

	first stage	development of the vehicle.
Cost	Number of stages	The cost is proportional to the number of stages that should be ad-hoc developed.
	Presence of propulsive system on each stage	The propulsive system is one the major component of cost for a vehicle. Different costs should be taken into and they are related in various ways to all the design phases and to the different maintenance and logistic activities to be carried out on ground.
	Presence of propellant tanks on each stage	The presence of propellant tanks on each stage can increase the cost due to the need of cross-feed and the deep impact on the additional maintenance activities that are required for the subsystems after each mission.
	Exploitation of existing first stage	The exploitation of already existing vehicles able to act as first stage can drastically reduce the cost of design and development of the transportation system. Considering the costs related to the operations of an existing first stage, the impact can be both positive or negative depending on the exploitation of a commercial aircraft or an expendable rocket.
Safety	Number of stages	The number of stages impacts on the safety, guaranteeing the possibility for the passenger compartment or the payload bay, to be separated from the rest of the transportation system. In reality, as it is detailed in Chapter 6, the single stage configuration can also be improved from the point of view of the safety, in different ways and the most promising one seems to be the design of a

		cabin escape system.
	Presence of propulsive system on each stage	The presence of a propulsive system in a stage guarantees additional manoeuvrability, enhancing the capability of surviving in case of emergency.
	Presence of propellant tanks on each stage	On the contrary with respect to the presence of a propulsive system, the on-board propellant is always considered a risk element for the passengers.

Starting from the qualitative analysis presented in Table 27, the following equations could be used in order to derive a first estimation of the FoMs with the aim of performing a high-level trade-off to derive the theoretical optimal staging strategy.

Complexity FoM

$$Complexity = C_{b1} n_{stages} \cdot \left[1 - (1-j)\left(\frac{1}{n_{stages}}\right)\right] + k_e \left(\sum_{i=1}^{n_{stages}} j_i e_i\right) + k_{t1}\left(\sum_{i=1}^{n_{stages}} t_i\right) + j k_{t2}\left(1 - \prod_{i=1}^{n_{stages}} t_i\right)$$

where:

C_{b1} is the basic level of complexity;

n_{stages} is the overall number of stages of the configuration;

i is an index representing each single stage;

e_i is a variable that indicates the presence of the propulsive system in the *i*-esim stage.

$e_i = 1$ means that the i-esim stage hosts a propulsive system

$e_i = 0$ means that the i-esim stage has not got a propulsive system

t_i is a variable that indicates the presence of the propellant system in *i*-esim stage.

$t_i = 1$ means that the i-esim stage hosts a propellant system

$t_i = 0$ means that the i-esim stage has not got a propellant system

k_e is a weighting factor that shows the impact of the propulsive system on the complexity FoM.

k_{t1} is a weighting factor that shows the impact of the propellant system on the complexity FoM.

k_{t2} is a weighting factor that shows the impact of the presence of cross-feed on the complexity FoM.

j is a *"switching"* variable that indicates the presence of already developed stages.

$j = 1$ means that all the stages should be properly designed and developed.

$j = 0$ means that the first stage is already existing.

j_i is a *"switching"* variable that indicates the presence of propulsion systems in already developed stages.

j_2 is always equal to 1 meaning that the second stage propulsion system should bead-hoc developed.

$j_1 = j = 0$ means that the propulsion system is related to an existing first stage.

$j_1 = j = 1$ means that the propulsion system is related to a first stage that should be developed yet.

Hypothesizing that the number of stages to be already developed is the real impacting factor on the Complexity, ($k_e = 0{,}2$ for single stage and $k_e = 0{,}8$ for two stages; $k_{t1} = 0{,}2; k_{t3} = 0{,}3$), the results show that the two two-stages configurations with an already developed first stage (Conf. 2.2 (a) and Conf. 2.4 (a)) are those characterized by the lowest complexity level. In particular, the best staging strategy to minimize the complexity Figure of Merit consists in a two-stages system with an already developed first stage and the simplest possible second stage, without any propulsion and propellant subsystems. This alternative is directly followed by the single stage architecture and by the other two-stages option with an already developed first stage and a second stage equipped with both a propulsion and a propellant system. On the contrary, the two stage configuration with both the stages to be developed from scratch appear to be the most complex.

Table 28: Evaluation of the FoM "Complexity"

	Single Stage	Conf. 2.2 (a)	Conf. 2.2 (b)	Conf. 2.4 (a)	Conf. 2.4 (b)	Conf. 3.2
C_{b1}	10	10	10	10	10	10
n_{stages}	1	2	2	2	2	2
j	1	0	1	0	1	1
k_e	0,2	0,8	0,8	0,8	0,8	0,8
e_1	1	1	1	1	1	0
e_{12}	0	1	1	0	0	1
j_1	1	0	1	0	1	1
j_2	1	1	1	1	1	1
k_{t1}	0,2	0,2	0,2	0,2	0,2	0,2
k_{t2}	0	0,3	0,3	0,3	0,3	0,3
t_1	1	1	1	1	1	1
t_2	0	1	1	0	0	1
Complexity	**10,4**	**11,2**	**22**	**10,2**	**21,3**	**21,2**

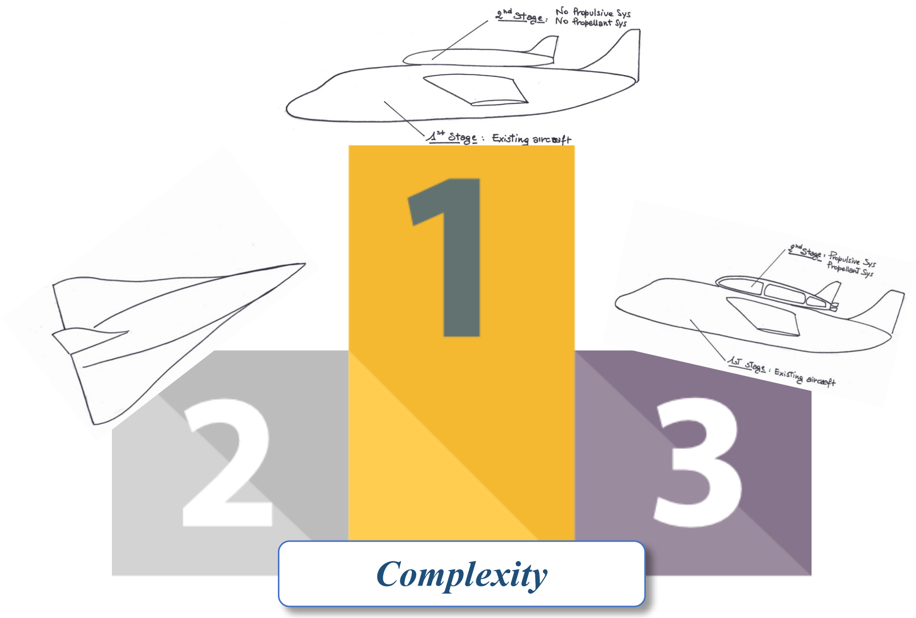


Figure 93: Pictorial representation of best Staging Strategy Alternatives with respect to the Complexity FoM.

Cost FoM

$$Cost = C_{b2} n_{stages} \cdot \left[1 - (1 - j)\left(\frac{1}{n_{stages}}\right)\right] + k_e \left(\sum_{i=1}^{n_{stages}} e_i\right) + k_{t1} \left(\sum_{i=1}^{n_{stages}} t_i\right)$$

where:

C_{b2} is the basic level of cost;

n_{stages} is the overall number of stages of the configuration;

i is an index representing each single stage;

e_i is a variable that indicates the presence of the propulsive system in the i-esim stage.

$e_i = 1$ means that the i-esim stage hosts a propulsive system

$e_i = 0$ means that the i-esim stage has not got a propulsive system

t_i is a variable that indicates the presence of the propellant system in i-esim stage.

$t_i = 1$ means that the i-esim stage hosts a propellant system

$t_i = 0$ means that the i-esim stage has not got a propellant system

k_e is a weighting factor that shows the impact of the propulsive system on the cost FoM.

k_{t1} is a weighting factor that shows the impact of the propellant system on the cost FoM.

j is a *"switching"* variable that indicates the presence of already developed stages.

$j = 1$ means that all the stages should be properly designed and developed.

$j = 0$ means that the first stage is already existing.

From the pure costs standpoint, it can be noticed that the single stage configuration results to be the most cost effective alternative followed by the two stages Conf. 2.4 (a) and Conf. 2.2 (a) with the already developed first stages.

Table 29: Evaluation of the FoM "Cost"

	Single Stage	**Conf. 2.2 (a)**	**Conf. 2.2 (b)**	**Conf. 2.4 (a)**	**Conf. 2.4 (b)**	**Conf. 3.2**
C_{b2}	10	10	10	10	10	10
n_{stages}	1	2	2	2	2	2
j	1	0	1	0	1	1
k_e	0,7	0,8	0,8	0,8	0,8	0,8
e_1	1	1	1	1	1	0
e_{12}	0	1	1	0	0	1
k_{t1}	0,1	0,1	0,1	0,1	0,1	0,1
t_1	1	1	1	1	1	1
t_2	0	1	1	0	0	1
Cost	**10,8**	**11,8**	**21,8**	**10,9**	**20,9**	**21**

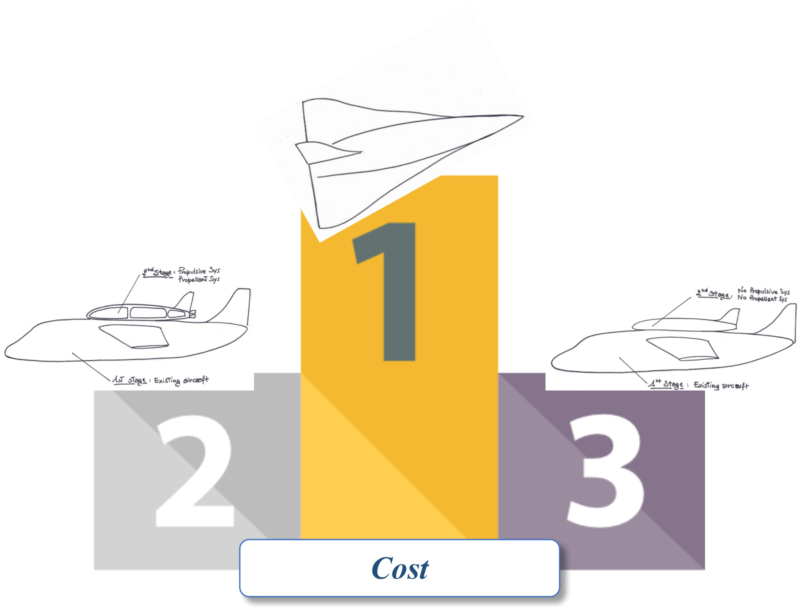


Figure 94: Pictorial representation of best Staging Strategy Alternatives with respect to the Cost FoM.

Safety FoM

$$Safety = S_b + \left(\sum_{i=1}^{n_{stages}} k_{ei} e_i \right) - k_{t1} \left(\sum_{i=1}^{n_{stages}} t_i \right)$$

S_b is the basic level of cost;

n_{stages} is the overall number of stages of the configuration;

i is an index representing each single stage;

e_i is a variable that indicates the presence of the propulsive system in the *i*-esim stage.

$e_i = 1$ means that the i-esim stage hosts a propulsive system

$e_i = 0$ means that the i-esim stage has not got a propulsive system

t_i is a variable that indicates the presence of the propellant system in *i*-esim stage.

$t_i = 1$ means that the i-esim stage hosts a propellant system

$t_i = 0$ means that the i-esim stage has not got a propellant system

k_{ei} is a weighting factor that shows the impact of the propulsive system of each single stage on the cost FoM.

k_{t1} is a weighting factor that shows the impact of the propellant system on the cost FoM.

Table 30: Evaluation of the FoM "Safety"

	Single Stage	**Conf. 2.2 (a)**	**Conf. 2.2 (b)**	**Conf. 2.4 (a)**	**Conf. 2.4 (b)**	**Conf. 3.2**
$\boldsymbol{S_b}$	10	10	10	10	10	10
$\boldsymbol{k_{e1}}$	1	0,1	0,1	0,1	0,1	0,1
$\boldsymbol{k_{e2}}$	0	0,9	0,9	0,9	0,9	0,9
$\boldsymbol{e_1}$	1	1	1	1	1	0
$\boldsymbol{e_{12}}$	0	1	1	0	0	1
$\boldsymbol{k_{t1}}$	0,1	0,1	0,1	0,1	0,1	0,1
$\boldsymbol{t_1}$	1	1	1	1	1	1
$\boldsymbol{t_2}$	0	1	1	0	0	1
Safety	**10,9**	**10**	**10**	**9,6**	**9,6**	**9,9**

Table 30 shows the results of the evaluation of Safety FoM for the different staging configurations, hypothesizing that the impact of the presence of propulsion system in the second stage on safety is noticeable (k_{e2} = 0,9). However, from the safety perspective, the single stage is considered to be most reliable especially thanks to the limited number of components. This configuration is followed by the two stage alternative with a first stage that acts as a propellant tank and by the two two-stages configurations in which the second stage is fully equipped with both engines and propellant.

Figure 95: Pictorial representation of best Staging Strategy Alternatives with respect to the Safety FoM.

Staging Strategy Trade-Off

The trade-off is carried out considering that all the three FoMs play a significant role in determining the optimal staging strategy. In particular, the following formulation can be adopted:

$$T.O. = \frac{K_1 \cdot safety}{K_2 \cdot complexity + K_3 \cdot cost}$$

Table 31: Trade Off results with $K_1 = K_2 = K_3 = 1/3$

	Single Stage	Conf. 2.2 (a)	Conf. 2.2 (b)	Conf. 2.4 (a)	Conf. 2.4 (b)	Conf. 3.2
Safety	10,9	10	10	9,6	9,6	9,9
Complexity	10,4	11,2	22	10,2	21,3	21,2
Cost	10,8	11,8	21,8	10,9	20,9	21
Trade-Off	**0,51**	**0,43**	**0,23**	**0,45**	**0,23**	**0,23**

Table 32: Trade Off results (sensitivity analysis)

	Weighting Factor				**Two Stages**				
	K_1	***K_2***	***K_3***	**Single Stage**	**Conf. 2.2 (a)**	**Conf. 2.2 (b)**	**Conf. 2.4 (a)**	**Conf. 2.4 (b)**	**Conf. 3.2**
Case 1	*0,33*	*0,33*	*0,33*	0,51	0,43	0,23	0,45	0,23	0,23
Case 2	*0,5*	*0,25*	*0,25*	1,03	0,87	0,46	0,91	0,45	0,47
Case 3	*0,25*	*0,5*	*0,25*	0,35	0,29	0,15	0,31	0,15	0,16
Case 4	*0,25*	*0,25*	*0,5*	0,34	0,29	0,15	0,3	0,15	0,16

Depending on the need or the performances expected by the stakeholders, the three FoMs can have a different impact on the selection of the optimal staging strategy. For this reason, the following table provides the results obtained for the basic case in which all the FoMs are supposed to have the same impact on the selection of the optimal solution. Moreover, in order to evaluate the consistency of the results with the variation of the weighting factors, different test cases have been carried out. The hypotheses about the weighting factors and the results are reported in Table 31. As it is possible to notice, the variation of weighting factors (Table 32) is not affecting the ordered list of the configuration, suggesting the two stages configuration exploiting an existing vehicle as first stage as the optimal staging strategy.

4.1.3 Propulsive Strategy

Referring to the observation done by H. J. Allen in 1958, "Progress in aeronautics has been brought about more by revolutionary than evolutionary changes in methods of propulsion" (Allen, 1958), it is easy to be understood that the propulsive strategy shall be properly investigated as soon as the mission profile has been defined. It is crystal clear that the selection of the most suitable propulsive system is strictly related to two major aspects of the mission profile: the operative environments and the maximum expected Mach number. In particular, in case of hypersonic and trans-atmospheric vehicles, due to wide range of speed regimes and the different operative environments that shall be considered within each single mission, an integrated propulsive strategy may be adopted, combining together different propulsive technologies to be exploited to operate the vehicle during the different mission phases. Taking a look at the current status of the propulsive technologies that could be exploited in the field of hypersonic and trans-atmospheric vehicles, it is possible to notice that both rocket and air-breathing propulsion systems may be adopted.

As far as rocket-based propulsion is concerned, liquid, hybrid and solid rocket may be employed. Complementary, looking at the more various world of air-breathing propulsion, both turbojet and turbofan can be theoretically exploited at the beginning of the mission profile, but they need to be supported by additional propulsion subsystems in order to reach the desired Mach numbers. In particular, Ramjet and Scramjet will be adopted. It is easy to be understood that depending on the Maximum achievable Mach number and the altitude at which a certain Mach number shall be reached, different propulsion subsystems will be exploited together. Figure 96 summarizes the major propulsive strategy alternatives.

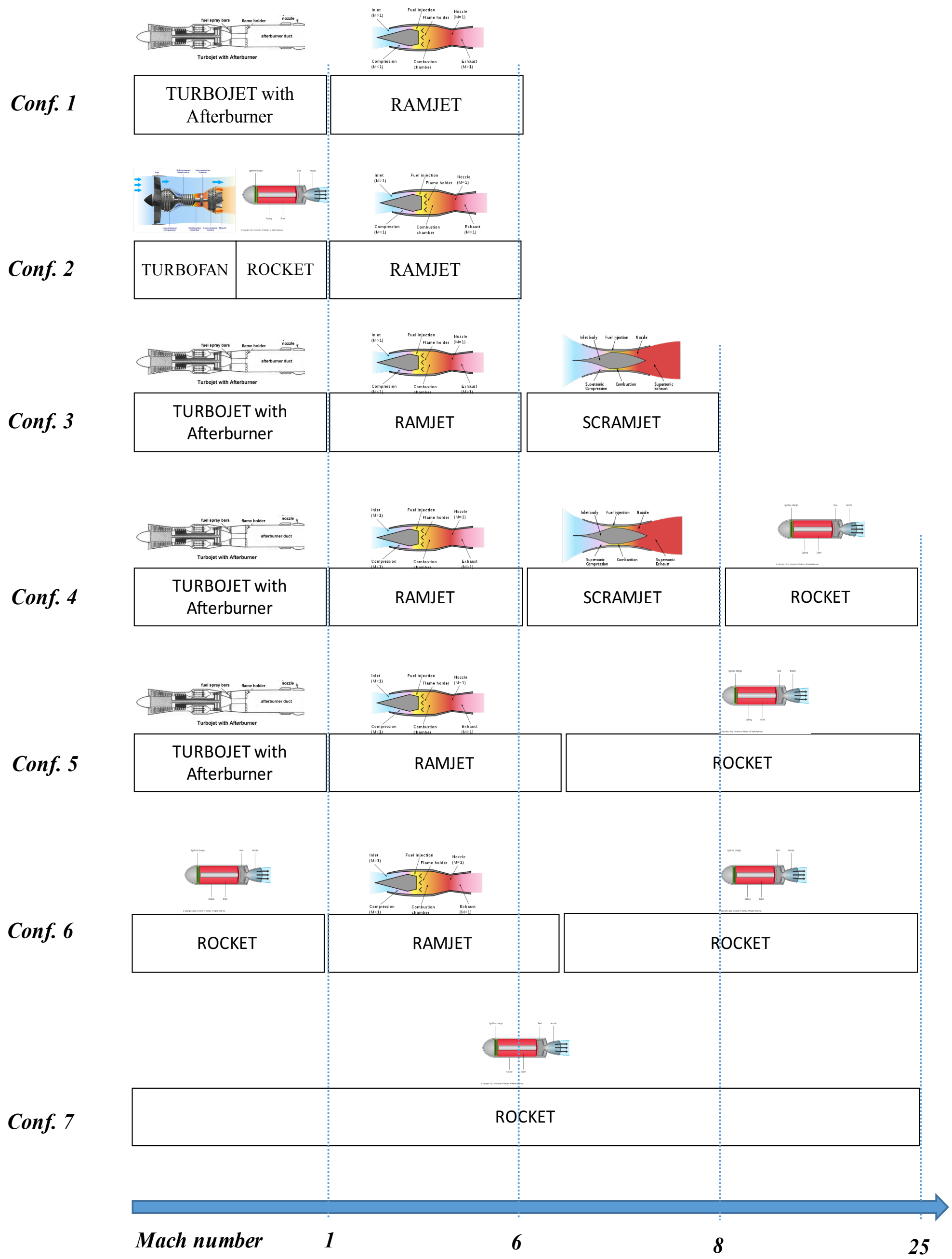


Figure 96: Propulsive Strategy Alternatives

It is worth noting that many currently under-development research activities in the field of hypersonic speed propulsion are focused on integrating within a single subsystem different propulsive technologies. Some of them have a long historical path coming back up to the Second World War. They are known as combined cycle or composite engines. Among the most relevant initiatives, it is useful to remember

- The Air Turbo Ramjet (ATR) a composite engine that behaves like a turbojet at very low speeds and as a rocket engine at higher speeds. Depending on the different applications, several variations on the theme have been developed, like:
    - the turbo ramjet rocket
    - the supercharged ejector ramjet (SERJ)
- The Dual Mode Ramjet (DMR) (Falempin, 2007) is a ramjet engine which can operate in both subsonic and supersonic combustion mode.
- Rocket Based Combined Cycle (RBCC)
- Turbine Based Combined Cycle (TBCC)

Other entirely separate classes of air-breathing engines specifically developed for the hypersonic application are the Liquid Air Cycle Engine (LACE) and the Inverse Cycle Engine. However, due to the relatively very low technology maturation these technologies cannot been currently exploited. However, future technological developments will provide the designer to include these propulsion systems within the set of eligible technologies.

The definition of the propulsion system shall be properly carried out selecting the best alternative for the different mission phases and trying to exploit the lowest number of different propulsive subsystems that can allow to fulfil all the mission requirements maximizing some Figures of Merit, such as cost, complexity and the overall vehicle mass (both dry and wet). The most recent research activities in the field of propulsive technologies for hypersonic vehicles are focusing on the integration of multiple propulsive subsystems into a single combined system with several different operative modes. Indeed, different types of combined propulsive cycles are currently under investigations.

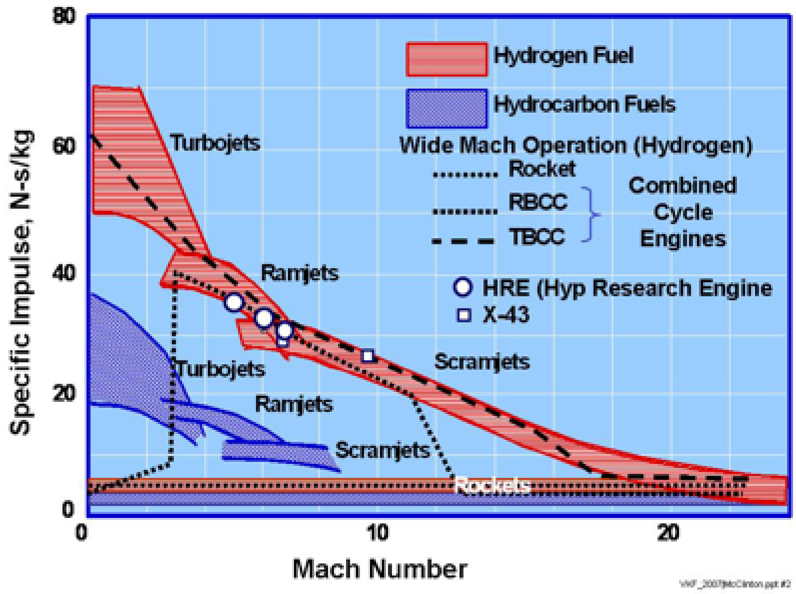


Figure 97: Specific Impulse vs Mach Number (McClinton, 2007)

It is clear that the selection of the proper propulsion system architecture can not only be perform on the basis of some qualitative considerations, but it is important to include some high level performances within the selection process. In particular, the minimum and the maximum achievable Mach number, the specific impulse, the thrust level and the current TRL should be properly considered. For example, it is possible to extrapolate useful data correlating specific impulse and Mach number for the different propulsive technologies. Moreover, two examples of under development propulsive systems, the Hypersonic Research Engine and the X-43 propulsion system have been inserted within the trends.

From the sketch in Figure 97, it is crystal clear that the current technology development status provides a lot of possible alternatives in terms of propulsive systems architecture. In particular, in order to rationalize the selection of the most suitable strategy, the following logical process has been followed:

1. Elicitation of the requirements with the highest impact on the propulsive strategy selection
2. Definition of the technical areas and technical aspects that will impact on the selection process.
3. Definition of the weighting criteria

4. Alternatives scoring process
5. Trade-Off

Starting from the analysis of the stakeholders' needs, it is possible to derive a first list of high level requirements, belonging to different areas, that could be used to derive at first, the areas of influence and then the most affecting characteristics. Looking to the wide range of possible hypersonic and trans-atmospheric missions, the following list of guide-requirements can be elicited. Please notice that this list aims to be as exhaustive as possible and it is not specific for a single type of mission. As it is clearly described in the case study paragraph, only a subset of requirements will be used for each specific mission design.

- The system shall be able to perform the take-off
- The system shall be able to perform a vertical the take-off
- The system shall be able to perform the climb in inner atmosphere
- The system shall be able to perform the climb in outer atmosphere
- The system shall be able to perform a subsonic cruise
- The system shall be able to perform a supersonic cruise
- The system shall be able to perform a hypersonic cruise
- The system shall be able to perform a powered re-entry phase
- The system shall be able to perform a powered cruise back to departure site
- The system shall be able to perform a powered descent
- The system shall be able to perform a powered landing
- The system shall be able to perform a vertical landing
- The effort required to carry out maintenance actions shall be reduced
- Thrust level shall be properly adjusted depending on the required in each mission phase
- Propulsion system shall be restart during the mission
- The system dry mass shall be minimized
- The system wet mass shall be minimized
- The mass of dangerous propellant shall be minimized
- Turn-around time shall be minimized
- Time-to-Market shall be minimized
- The overall system costs shall be minimized
    - The propulsive system research and development costs shall be minimized

- The propulsive system operational costs shall be minimized
- The propulsive system test and verification costs shall be minimized

From this list of requirements, it is possible to notice that the research areas that would have a greater impact on the propulsion strategy selection will be weight and balance, operations, and maintenance. Table 33 aims at summarizing the major reasons why these areas will impact on the propulsion strategy selection, while Table 34 suggests scoring strategy for the selection of the optimal propulsion strategy. Please note that in case it is not possible to numerically evaluate some parameters, a proper scale has been used to translate qualitative evaluation (L = low; M = medium; H = high) in numerical evaluations, to be exploited within the ad-hoc built equations.

Table 33: Areas of Interests impacting on Propulsive strategy alternative selection

Areas of Interest	Drivers	Impact on the propulsion strategy selection
Weight and Balance	• Number of different propulsion systems	The highest is the number of different propulsion systems, the highest the dry mass associated to the overall dry mass. In case of different propulsion systems fully integrated within a single propulsive element, a reduction factor may be considered. [REF]
	• Wall temperature	The wall temperature is an optimal indicator of the mass increase due to the need of active cooling and thermal protection systems.
	• Presence of rotating machinery	The presence of rotating machinery is undoubtedly contributing to increase the mass and the complexity of the overall vehicle.
	• Presence of oxidizer	The need of carrying proper oxidizer on-board, increases the overall mass of the vehicle, affecting both the dry and the wet mass.

Operations	• Re-start capability	The possibility for a propulsion system to be restarted allows to enlarge the operative scenarios.
	• Throttling capability	The possibility of playing with the thrust module allows to enlarge the ranges of application of this propulsive system
	• Maximum Operative Mach number	The maximum operative Mach Number defines the possibility of exploiting a certain propulsive system in each single mission phases.
	• Thrust Vectoring capability	The possibility of guaranteeing a Thrust Vectoring allows perform vertical/short take-off and landing
Maintenance	• Number of different propulsion systems	The highest is the number of different propulsion systems, the highest will be the maintenance actions required. In case of a highly integrated solutions, this value can also be increased. In addition, it increases the need of additional specialized technicians to carry out the maintenance actions.
	• Wall Temperature	The wall temperature is an indicator of the criticalities that characterize propulsion system structure and material. Indeed, the highest is the wall temperature, the heaviest will be the required maintenance actions.
	• Presence of rotating machinery	The presence of rotating machinery increases diminishes the reliability of the system, theoretically. In order to keep it constant, additional maintenance actions will be required.
	• Propellant type	The type of propellant used by the several different types of engines increases the need of additional specialized technicians to carry out the maintenance actions. Moreover, the maintenance actions will be required more frequently. However,

		this is a more detailed choice that could be perform later on in the design process.
	• Presence of Oxidizer	The presence of on-board oxidizer will require additional maintenance actions

Table 34: Score assignments for the different propulsive strategies

	Drivers	**C1**	**C2**	**C3**	**C4**	**C5**	**C6**	**C7**	**C8**
		TJ with AB Ramjet	*TF Rocket Ramjet*	*TJ with AB Ramjet Scramjet*	*TJ with AB Ramjet Scramjet Rocket*	*TJ with AB Ramjet Rocket*	*Rocket Ramjet Rocket*	*TJ with AB Rocket*	*Rocket*
Weight and Balance	[A1] Number of different propulsion systems	2	3	3	4	3	3	2	1
	[A2] Wall temperature	M(5)	H(10)	H(10)	H(10)	M(5)	M(5)	M(5)	M(5)
	[A3] Presence of rotating machinery	1/2	1/3	1/3	1/4	1/3	0	1/2	0
	[A4] Presence of oxidizer	1/2	1/3	0	1/4	1/3	2/3	1/2	1/1
Operations	[B1] Re-start capability	M(5)	L(3)	M(5)	M(5)	M(5)	L(3)	M(5)	V(1)
	[B2] Throttling capability	M(5)	L(3)	M(5)	M(5)	M(5)	L(3)	M(5)	V(1)

	[B3] Maximum Operative Mach number	6	6	8	25	25	25	25	25
	[B4] Thrust Vectoring capability	Y(1)	N(0)	Y(1)	Y(1)	Y(1)	N(0)	Y(1)	N(0)
Maintenance	[C1] Number of different propulsion systems	2	3	3	4	3	3	2	1
	[C2] Wall Temperature	M(5)	H(10)	H(10)	H(10)	M(5)	M(5)	M(5)	M(5)
	[C3] Presence of rotating machinery	1/2	1/3	1/3	1/4	1/3	0	1/2	0
	[C4] Presence of Oxidizer	1/2	1/3	0	1/4	1/3	2/3	1/2	1/1

In order to evaluate the best alternative in terms of propulsive strategy, the different Figures of Merit listed in the previous table have been combined as follows:

$$TO = -K_A * \sum (A_i)_n + K_B * (B_i)_n - K_C * \sum (C_i)_n$$

where

TO is the global FoM

K_A is the weighting factor taking into account the impact of weight & balance area of interest on the selection of the propulsive strategy. The minus sign is due to the fact that the characteristics afferent to this area of interest are playing against it.

K_B is the weighting factor taking into account the impact of maintenance area of interest on the selection of the propulsive strategy.

K_C is the weighting factor taking into account the impact of operations area of interest on the selection of the propulsive strategy. The minus sign is due to the fact that the characteristics afferent to this area of interest are playing against it.

$(A_i)_n$ is the normalized estimation obtained as $\frac{A_i}{\max(A_i)}$.

$(B_i)_n$ is the normalized estimation obtained as $\frac{B_i}{\max(B_i)}$.

$(C_i)_n$ is the normalized estimation obtained as $\frac{C_i}{\max(C_i)}$.

Table 35: Sensitivity Analysis for the propulsive strategy selection

			C1	**C2**	**C3**	**C4**	**C5**	**C6**	**C7**	**C8**
K_A	**K_B**	**K_C**	*TJ with AB Rocket*	*TF Rocket Ramjet*	*TJ with AB Ramjet Scramjet*	*TJ with AB Ramjet Scramjet Rocket*	*TJ with AB Ramjet Rocket*	*Rocket Ramjet Rocket*	*TJ with AB Rocket*	*Rocket*
1/3	***1/3***	***1/3***	-0,59	-1,35	-0,64	-0,63	-0,17	-0,54	-0,33	-0,70
1/2	***1/4***	***1/4***	-1,07	-1,70	-1,08	-1,16	-0,69	-0,89	-0,88	-0,96
1/4	***1/2***	***1/4***	0,37	-0,66	0,25	0,43	0,88	0,14	0,75	-0,18
1/4	***1/4***	***1/2***	-1,07	-1,70	-1,08	-1,16	-0,69	-0,89	-0,88	-0,96

As it is possible to notice in Table 35 different weighting strategies have been tested, performing a sensitivity analysis of the results. The solution provides to be robust enough. Indeed, the configuration with the highest scoring results is always the configuration that exploits in series a Turbojet with afterburner, a ramjet and a rocket technology. Depending on the weighting strategy adopted, the second and the third suggested strategies may vary.

It is clear that, depending on the specific case study, some tuning of the inserted values should be performed. In particular, as it will be clearly demonstrated with the help of the case-study, some high level stakeholders' requirements or other high level mission constraints can prevent the designer to consider one or more of the proposed configurations.

4.1.4 Take-off and Landing Strategy

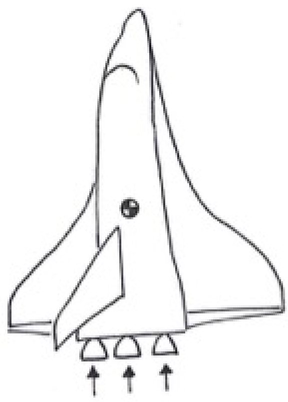
The take-off and landing strategy are other elements with a deep impact on the overall configuration of the vehicle as it will be discussed in the chapter dealing with integration. However, in this context, without focusing on the different technologies that could allow the different strategies, the author aims at providing the reader with an overview of the take-off and landing alternatives and the connected impact on vehicle and spaceport infrastructures.

The most conventional take-off strategy is the horizontal one from a prepared runway. In this case the transportation system should be equipped with air-breathing engines able to provide the thrust required to fulfil the take-off requirement. The strategy combining a horizontal take-off and a horizontal landing is the less impacting from the point of view of the infrastructures that should be developed to support the mission. In addition, the vehicle can be theoretically operated in many different locations (the list of theoretical feasible sites should be pruned of the locations not respecting the safety parameters) enhancing the possibility of increasing economical return.

The case of vertical take-off is more difficult to be executed from the technical standpoint. Two different alternatives could be envisaged: a tail sitting and a non-tail-sitting solution. The tail-sitting solution, multiple stages transportation systems, especially those exploiting launch vehicles, is typical of rocket propelled first stage. In this case, the take-off strategy deeply impacts on the on-ground infrastructure, requiring a proper launch pad and tower and all the logistics related to this selection. This strategy can be coupled with both a vertical (controlled or simple splash down) or a horizontal landing (Space Shuttle like). In case the transportation system aims at accommodating passengers, in particular if they are not trained, a non-tail-sitting solution would be preferable both in terms of accessibility and comfort (considering the g-force direction during the take-off phase). Unfortunately, this strategy, for both take-off and landing, implies technical difficulties, forcing the use of heavy solutions like those exploited in military aircraft like Harrier. Conversely, this strategy diminishes the impact on the spaceport theoretically allowing operations from which ever semi-prepared location. At this high-level of detail, it is not convenient to carry out a trade-off analysis as in the case of the staging strategy because the selection of the optimal propulsive strategy would be closely related to the different systems that would be installed on the vehicle.

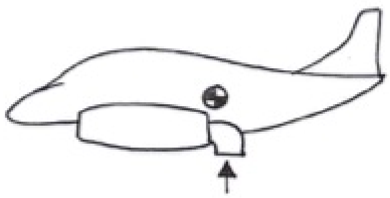
Considering the vertical take-off strategy, different system architectures could be hypothesized, especially in case of exploitation of air-breathing engines. The following table aims at summarizing the most promising system architecture alternatives figuring out whether or not they are exploitable in hypersonic transportation systems. In particular, from this preliminary analysis, it is clear that the configuration for the take-off can have a deep influence on accessibility, comfort and complexity and these parameters could be used for the trade-off. In general, considering that modern jet fighters have a thrust-to-weight (T/W) ratio that exceeds 1, the easiest way to carry out a vertical take-off seems to simply point the exhaust gas downward but unfortunately some problems may arise. In particular, the most evident problem is emerging when attempting to deflect the thrust, since an unbalanced configuration appears. According to Raymer (Raymer, 2012), the balance problem is possibly the single most important driver of the design of VTOL vehicles. As it is clearly in Table 36 two main options could be envisaged to overcome this problem. Either the thrust can be moved in a position that is closer to the Centre of Gravity (CG) or alternatively, an additional thrust force can be located near the nose. It is clear that both the strategies have impact on the layout of the vehicle and represent a compromise among different factors (aerodynamic efficiency, additional weight, costs, complexity, etc.). A second issue that should be properly taken into account is the problem of thrust matching that affects all the vehicles that implies a VTOL strategy, since they have much lower thrust demand in cruise. In case the designers decide to optimize the engine for the cruise requirement, the over-boost required to allow the vertical take-off should be generated separately, by means of “lift-engines”. For the sake of clarity, exploiting the same nomenclature proposed in (Raymer, 2012), the author will use the term L+L/C indicating a configuration consisting of engines devoted to generating Lift force during take-off (L) and the main engine (L/C) that is also used during take-off but it is optimized for the cruise segment. In the following table, other intermediate configurations are proposed and the possibility of their exploitation for hypersonic transportation systems is discussed. In particular, the first three architectural alternatives propose the exploitation of conventional engines without flow diversion. The forth, the fifth and the sixth configurations suggest the diversion of the exhaust gasses while the last alternatives are characterized by the presence of lift engines. Different alternatives could be obtained modifying the structure of the engine, based upon the concept of split flow, like exploited in the case of Pegasus, the main engine of Harrier AV-8. In this case, the fan air and the core air are each separately vectored through elbow nozzles simplifying the transition between vertical and forward flight and enhancing manoeuvrability. On the negative side, this kind of configuration suffers the thrust matching problem. Moreover, due to the location that should be closed enough to the CG, the wave drag can drastically increase due to the increment of the cross-sectional area, at the fuselage-wing location, preventing high speed flight.

Table 36: Architectures for vertical take-off strategy

Vertical Take-off strategy	***Description***
Tail sitting *(Conf. 1)* 	The vertical tail sitting configuration is the easiest way to perform a vertical take-off and it is the one exploited in many space applications, were rocket propulsion is used since the beginning of the mission. This strategy has a low impact on the architecture of the vehicle but requires ad-hoc on-ground infrastructures. In addition, this strategy is not the best from the passengers' standpoint. Indeed, a part from the logistic problems related to the accessibility of the crew compartment, the position of the passengers during the very first moments of the mission is not optimal. The problem could be overcome through the exploitation of gyroscopic seats

Vectored Thrust at CG

(Conf. 2)



Basic Configuration:

This configuration is characterized by the position of the engine that is placed in a location that allows the thrust to be vectored downward in correspondence of the CG location.

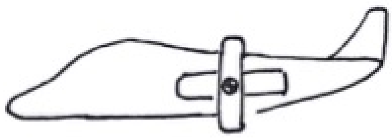
Advanced Configuration:

There could also be the possibility of having vectoring nozzles at the centre of gravity with the engines out in front like the Yak 36 or the X-14 with several drawbacks in terms of pilot visibility and thermal and acoustic problems. There is also another possible alternative called RIVET (Reverse Installation Vectored Engine Thrust). This is an alternative approach that places the nozzles at the centre of gravity but puts the engine in the rear fuselage with a backward installation. This type of architecture is characterized by a lower level of complexity, reduced weight, ease of transition between vertical and horizontal flight and an inherent vectoring in forward flight. Advantages of this configuration have been demonstrated in (Raymer, 1990) (Raymer, 1988) and the solution appears viable but an overall inlet duct loss of about 5% should be taken into account, in view of the presence of a 180 degrees bend required to supply air to a backward-mounted engine. Although the evident advantages in terms of performance of these configurations, there is only a very limited possibility of implementing them in a hypersonic transportation vehicle.

The possibility of mounting tilt nacelles has been envisaged in the past, mainly for naval applications. Unfortunately, this strategy is characterized by a high level of complexity especially due to the mechanism that should be designed and implemented to allow the movement of the

Tilt Nacelle at CG

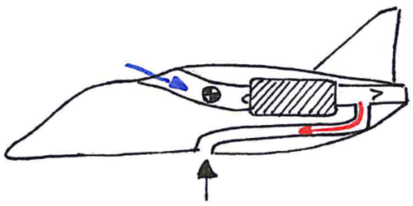
(Conf. 3)



nacelles. Moreover, this strategy can only be applied in case of engines mounted externally and mainly for this reason it cannot be considered viable for a hypersonic transportation system.

Unaugmented Flow

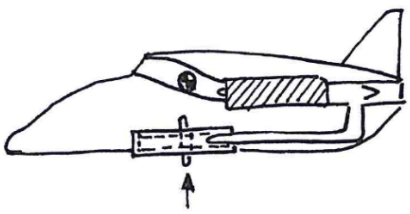
(Conf. 4)



This configuration exploits the diverted flow exhausting it directly downwards without any intermediate devices. In this case, the precision of the thrust vectorization only depends on the precision of the outlet duct that should end in correspondence to the CG location in order to guarantee the required stability. This configuration is a good compromise between complexity and weight. The only problem is that in order to fulfil the VTOL requirement, the engine results to be oversized for the cruise but this can be irrelevant for the case of parabolic flights where the air-breathing engines are only used in the very beginning of the mission and the thrust required to reach the target altitude can be more easily supplied by rocket engines.

Tip-driven fan

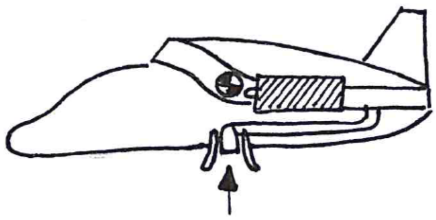
(Conf. 5)



The tip-driven fan configuration is very similar to the previous one but there is a gas driven fan that is a ducted fan turned by turbine blades spun by diverted engine exhausts or, in some cases, diverted compressor air. The Ryan XV-5A used tip-driven fan, attaining an augmentation ratio of almost 3, resulting to be the highest augmentation ratio ever obtained for a jet VTOL. Unfortunately, this configuration is not optimal for vehicles aimed at performing an atmospheric re-entry. Indeed, the presence of a fan causes an interruption of the structure of the spaceplane airframe, causing a weight increment due to the need of further structural reinforces. Moreover, in order to allow a re-entry phase, a proper device should be envisaged to cover the hole in the structure with proper materials allowing thermal protection. Conversely, this configuration can allow to reduce the problem of thrust matching between take off and cruise phase and could be highly recommended for all those missions in which the cruise represents the core of the flight profile and thus, should be optimized.

Ejectors

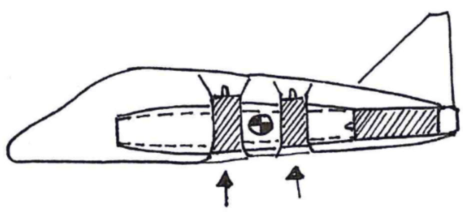
(Conf. 6)



This configuration is very similar to the previous two but it uses ejectors in order to allow thrust augmentation. The ejector makes use of viscosity of the air. Any exhaust jet will drag along adjacent air molecules, accelerating the free air in its vicinity. While ejectors promise theoretical augmentation ratios of 3 or more, a more realistic range of 1,3 – 2,2 can be envisaged [RAy]. Like tip-driven fan, also this configuration tends to chop up the structure of the vehicle but considering the dimensions of the structural interruptions, in this case, the replacement with ad-hoc designed sliding doors could be easily implemented. Also in this case, the ejectors allow to reduce the thrust matching problem. In addition, the ejectors, if properly designed, can allow a certain steering capability. In this case, the transition between vertical and horizontal flight can also be reduced.

Separate Lift Engines

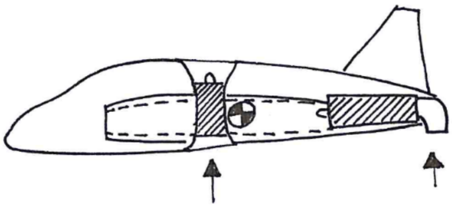
(Conf. 7)



Despite of the additional weight and volume due to the presence of separated lift engines, this configuration has been adopted in famous aircraft, like for example the Mirage III-V because able to overcome the thrust-matching problem completely. According to (Raymer, 2012) future lift engines could be so innovative to allow a T/W of about 25. In case of doors and vectoring nozzles, the installation cannot be neglected (the weight can roughly double). The presence of a lift engine in front of the CG, just behind the cabin, cannot be compatible with the presence of a cabin compartment. In addition, splitting the thrust vector in two components in front and behind the CG enhance the stability in nominal conditions but can cause instability problems in case of failure.

L+L/C vectored

(Conf. 8)



Basic Configuration:

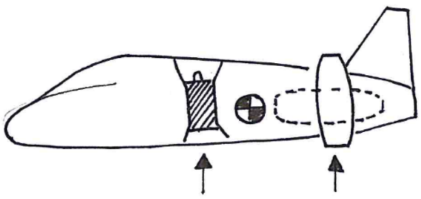
The L + L/C configuration suffers of the same safety problem of the previous configuration but allows to save mass because it exploits the main engine also for the take-off using steerable nozzles. In order to ease the transition between vertical and horizontal flight also the lift engine can be endowed of steering capabilities.

Advanced Configuration

An advanced configuration exploiting this kind of architecture is the shaft driven lift fan, currently adopted by the F-35B. In this case, the front lift engine is replaced with a fan which is really more similar to a jet engine compressor than a shrouded propeller. This fan is mechanically spun by a driveshaft that is powered by an extra turbine in the engine exhaust.

L+L/C with tilt Nacelles

(Conf. 9)



This configuration is really complex and implies a noticeable increment in terms of mass. Besides the easier transition between vertical and forward flight, this configuration cannot be easily implemented starting from the necessity of having enteral mounted engines.

In order to guide engineers through the selection of the most suitable take-off strategy, an approach very similar to the one suggested and applied for the selection of the propulsive alternative, has been considered. In particular, starting from the analysis of the stakeholders, the following list of requirements have been considered to have a certain impact on the selection of the take-off strategy to be adopted.

- Passengers shall be properly accommodated within the cabin and proper comfort level shall be guaranteed during the overall mission profile.
- Payload shall be properly accommodated during the overall mission profile
- Non trained passengers should be safely hosted
- The systems acceleration shall be within the limits
- Pilot visibility shall be properly guaranteed during the overall mission profile
- Propulsion system dry mass shall be minimized
- Propulsion system wet mass shall be minimized
- System structural mass shall be minimized
- On-ground personnel safety shall be maximized
- The risk associated to third-parties' injuries shall be minimized
- On-board passengers and crew safety shall be guaranteed
- The transportation system shall be able to perform a vertical take-off from un-prepared fields
- The transportation system shall be able to perform a vertical landing to un-prepared fields
- Stability shall be maximized during take-off and landing phases
- The transition from vertical to horizontal flight shall be properly guaranteed.
- Turn-around time shall be reduced
- Maintenance efforts shall be reduced
- The transportation system shall withstand the thermal loads during the overall mission phases
- The transportation system drag coefficient shall be reduced.
- The impact of take-off and landing operations on the on-ground facilities shall be minimized

Then, Table 37 summarizes the major areas of interest for this trade-off and for each of them, a list of drivers is presented.

Table 37: Areas of Interest impacting on the selection of the best architecture for take-off and landing strategy

Areas of Interest	Drivers	Impact on the take-off and landing strategy selection
Passengers and Payload accommodation	• Passengers accommodation	Depending on the take-off and landing strategy that is adopted, passengers can experience different level of comfort, due to the seat position with respect to the horizontal direction.
	• Payload accommodation	In case a scientific payload is included, it shall withstand proper loading limits.
	• Maximum acceleration	The need of hosting non-trained passengers on-board would prevent the designer to adopt too hard vertical take-off and landing solutions
Aerodynamic and aero-thermodynamic	• Maximum heat load	The strategy adopted to perform the take-off and landing strategy can make use of additional devises that would increase the overall heat loads. This problem can be reduced with retractable devices or proper thermal protection strategy.
	• Drag coefficient increase	External devises used to allow a vertical take-off and landing activities shall be minimized to avoid excessive increment in aerodynamic drag.
Structure	• Structural Reinforcements required	The presence of tilting or movable devices may imply structural interruptions. This will require additional reinforcements and thus an increase in of the transportation

		system dry mass
Operations	• Pilot visibility	Depending on the selected take-off and landing strategy, the pilot visibility shall dramatically be reduced.
	• Take-off from (landing in) un-prepared field	The strategy selected to perform the vertical take-off and landing can have a different impact on the possibility of being operated from un-prepared fields. In some cases, there would also be the possibility. It is strictly related to the on-ground clearance.
	• Turn-around Time	Depending on the complexity of selected strategy, and related propulsive subsystems, the turn-around time can be deeply affected
Stability and Control	• Stability during vertical flight	The number of thrust points will define the level of stability of the vehicle during take-off and transition phase.
	• Control during vertical flight	The controllability during the most critical phases of the mission profile will be investigated in detail. However, at this stage, it is possible to understand if the selected strategy requires special control laws to be controlled, especially during the transition phase.
	• Transition from vertical to horizontal flight	The strategy shall allow a proper transition phase, ensuring proper, comfort and safety levels.
Logistic and Maintenance	• Maintenance	The systems adopted to carry out the selected take-off and landing strategy can be properly identified in order to reduce the required maintenance actions.
	• Proper facilities	The need of proper facilities required to support the take-off and landing operations may be

		detrimental for mission costs and complexity
Safety	• Passengers and crew safety • On-ground personnel safety • Third-parties' safety	The strategy will have an impact on safety at different levels. However, the impact on Safety is very difficult to be estimated at this level. Additional details about the subsystem architecture must be known.

In addition, all the Drivers have been evaluated for all the configurations presented in Table 38:

Table 38: Scoring strategy for the selection of the best strategy to perform vertical take-off and landing

	Drivers	**C1**	**C2**	**C3**	**C4**	**C5**	**C6**	**C7**	**C8**	**C9**
		Tail Sitting	*Vectored Trust at CG*	*Tilt Nacelle at CG*	*Un-augmented flow*	*Tip driven fan*	*Ejectors*	*Separate Lift Engines*	*L+L/C vectored*	*L+L/C tilt nacelles*
Accommodation	[A1] Passengers accommodation	L(3)	M(7)	M(7)	H(10)	H(10)	M(7)	H(10)	L(3)	L(3)
	[A2] Payload accommodation	L(3)	M(7)	M(7)	H(10)	H(10)	M(7)	H(10)	L(3)	L(3)
Aerodynamic and aero-thermodynamic	[B1] Maximum acceleration	H(10)	M(7)	M(7)	M(7)	L(3)	M(7)	L(3)	H(10)	H(10)
	[B2] Maximum heat load	0	M(7)	M(7)	L(3)	H(10)	M(7)	M(7)	L(3)	M(7)

	[B3] Drag coefficient increase	0	M(7)	H(10)	L(3)	M(7)	L(3)	L(3)	L(3)	H(10)
Structure	[C1] Structural Reinforcements required	0	0	M(7)	M(7)	H(10)	M(7)	H(10)	H(10)	H(10)
Operations	[D1] Pilot visibility	H(10)	M(7)	M(7)	M(7)	M(7)	M(7)	M(7)	M(7)	M(7)
	[D2] Take-off from (landing in) un-prepared field	Y(1)	Y(1)	Y(1)	Y(1)	N(0)	Y(1)	N(0)	N(0)	N(0)
	[D3] Turn-around Time	H(10)	L(3)	M(7)	M(7)	H(10)	M(7)	M(7)	M(7)	H(10)
Stability and Control	[E1] Stability during vertical flight	M(7)	L(3)	L(3)	M(7)	H(10)	M(7)	H(10)	H(10)	H(10)
	[E2] Control during vertical flight	L(3)	M(7)	M(7)	L(3)	L(3)	M(7)	L(3)	M(7)	H(10)
	[E3] Transition from vertical to horizontal flight	L(3)	H(10)	H(10)	M(7)	M(7)	M(7)	M(7)	H(10)	H(10)
Logistic and Maintenance	[F1] Maintenance	H(10)	M(7)	M(7)	M(7)	H(10)	M(7)	M(7)	M(7)	H(10)
	[F2] Proper facilities	H(10)	L(3)	L(3)	L(3)	L(3)	L(3)	L(3)	L(3)	L(3)

Table 39: Sensitivity Analysis for the selection of the best take-off and landing strategy

Weighting Factor						C1	C2	C3	C4	C5	C6	C7	C8	C9
K_A	**K_B**	**K_C**	**K_D**	**K_E**	**K_F**	*Tail Sitting*	*Vectored Trust at CG*	*Tilt Nacelle at CG*	*Un-augmented flow*	*Tip driven fan*	*Ejectors*	*Separate Lift Engines*	*L+L/C vectored*	*L+L/C tilt nacelles*
1/6	*1/6*	*1/6*	*1/6*	*1/6*	*1/6*	-0,68	-0,28	-0,52	-0,28	-0,33	-0,38	-0,12	-0,28	-0,52
1/2	*1/10*	*1/10*	*1/10*	*1/10*	*1/10*	-0,17	0,39	0,25	0,63	0,60	0,33	0,73	0,07	-0,07
1/10	*1/2*	*1/10*	*1/10*	*1/10*	*1/10*	-0,81	-1,01	-1,27	-0,69	-1,00	-0,91	-0,59	-0,81	-1,39
1/10	*1/10*	*1/2*	*1/10*	*1/10*	*1/10*	-0,41	-0,17	-0,59	-0,45	-0,60	-0,51	-0,47	-0,57	-0,71
1/10	*1/10*	*1/10*	*1/2*	*1/10*	*1/10*	-1,61	-0,97	-1,27	-1,13	-0,88	-1,19	-0,63	-0,73	-0,99
1/10	*1/10*	*1/10*	*1/10*	*1/2*	*1/10*	0,11	0,63	0,49	0,51	0,60	0,61	0,73	0,91	0,89
1/10	*1/10*	*1/10*	*1/10*	*1/10*	*1/2*	-1,21	-0,57	-0,71	-0,57	-0,72	-0,63	-0,47	-0,57	-0,83

From the results reported in Table 38 and Table 39 it is clear that the configuration envisaging separate engines properly dedicated for the take-off and landing phases appears to be the most promising. However, this is one of those architectural configurations that could be not applicable in depending on structural and aerothermodynamic requirements. In any case, the suggested methodology can be applied adopting different scoring scale or simply changing the relative weight of the components of the FoM.

4.1.5 Aerothermodynamics Configuration

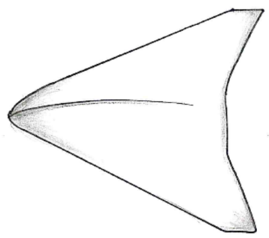
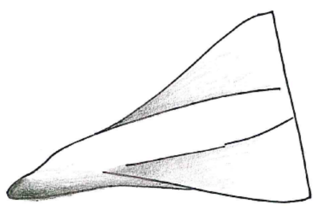
Each stage can have its proper configuration and it is important but not an easy task to establish high-level classifications. It is easier to hypothesize that several different criteria may be adopted for this purpose. In particular, considering the classification criteria already available in textbooks, Lift-over-Drag ratio (L/D), the ratio between the two main dimensions of the object (d1/d2) and the presence or absence of wing (that can affect the L/D parameter) can be considered as the major features affecting the configuration of each single stage.

At very high level, the configurations can be easily divided in the following categories, already announced at the beginning of the chapter:

- Re-entry Vehicles (RV)
- Ascent and re-entry vehicles (ARV)
  - Orbital ascent and re-entry vehicles (O-ARV)
  - Suborbital ascent and re-entry vehicles(SO-ARV)
- Cruise and acceleration vehicles with air-breathing propulsion (CAV)

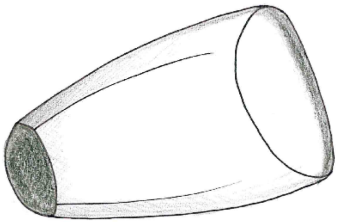
For each of these families of transportation system architectures, different aero-thermodynamic configurations can be envisaged, as highlighted in the following Tables. In addition to a brief description of each configuration, an indication of the L/D parameter and of existing or under-development projects are reported.

Table 40: Possible Aerothermodynamic configurations

Possible aero-thermodynamic configuration	**L/D**	**Description**
Winged Vehicle (Flying Wing) *Conf. 1* 	L/D > 1.4	This configuration is characterized by a wide lifting surface. The fuselage is not clearly distinct with respect to the wing. This configuration maximizes the range in (powered or unpowered) gliding phases. This configuration is preferable in case a high number of passengers should be hosted. In this case, bubble structures can be exploited to maximize the volumetric efficiency minimizing the impact on weight. NASA is currently focusing on this kind of configuration, with the idea of resuming the heritage of some X-series projects such as the X-33. This configuration can be theoretically exploited for RVs, ,O-ARVs, SO-ARVs and CAVs.
Winged-Re-entry Vehicle (Fuselage + wing) *Conf. 2* 		The winged vehicle is the most traditional configuration, where fuselage and wing are clearly separated. In this case, the passengers compartment is hosted within the available room in the conical section, while the wing surface is the major responsible for the aerodynamic lift generation. This configuration can be a good compromise among several mission needs. X-38, Phoenix and Hope demonstrators are examples of implementation of a winged configuration for a re-entry mission. This configuration can be theoretically exploited for RVs, O-ARVs, SO-ARVs and CAVs.

Non -Winged-Re-entry Vehicle (Lifting Body)

Conf. 3

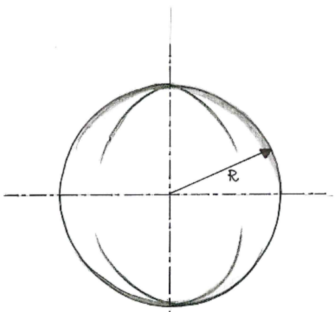


L/D = 1 – 1.4

The lifting body configuration can be considered optimal as far as the capability of withstanding thermal loads during re-entry is concerned. On the contrary, special Guidance Navigation and Control systems should be envisaged in order to enhance the controllability of this object. In addition, it is worth noting that this configuration is more suitable in case a limited amount of payload should be transported. This architecture has been exploited for the ESA IXV project and it is a candidate for the Space Rider vehicle. In addition to orbital re-entry, this aero-thermodynamic configuration could be envisaged for O-ARVs and SO-ARVs. Due to its aerodynamic characteristics, this configuration can hardly be exploited to perform an autonomous ascent or cruise phase.

Spherical Capsule

Conf. 4

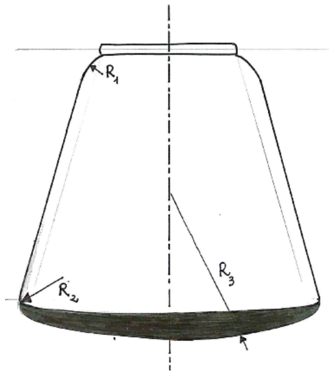


L/D = 0

The spherical shape is the simplest configuration that can be envisaged to perform a re-entry. Considering the impossibility of including flight control systems within this configuration, the spherical shape is the worst in terms of controllability and manoeuvrability. On the opposite, it can provide high volumetric efficiency with optimal weight distribution. Suitable for small number of passengers. This type of configuration, as well as all the other capsule-like configurations, are suitable for re-entry missions only.

Blunt Cone Capsule

Conf. 5



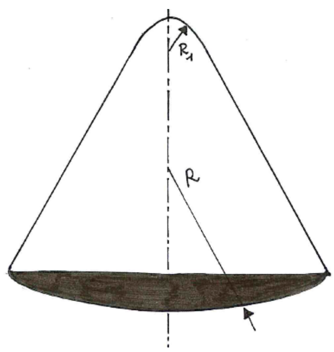
L/D = 0.35 – 0.45

The blunt cone capsule is the best compromise among the aero-thermodynamic efficiency, simplicity and the volumetric efficiency. Indeed, the shape allows to make a clear division of the available volume, in a lower part, in which the major subsystems could be located and the upper part for the passengers.

This shape has been exploited by the Russian since the 1965 especially for Low Earth Orbit missions. SOYUZ capsules are clear examples of blunt cone shape capsules.

Conic Capsule

Conf. 6



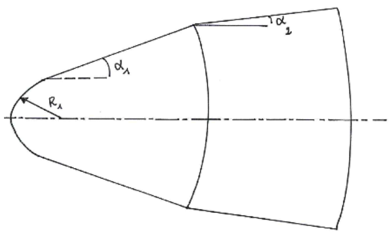
L/D = 0.35 – 0.4

The conic capsule could be considered as the most aerodynamically efficient capsule and this is mainly due to the differences in the radius of the upper and lower part. From the stability point of view, this configuration has the advantages of an axisymmetric shape.

Several vehicles have been developed and manufactured in the USA. In particular, during the APOLLO project, different models flown during 1966 – 1973 period.

Bluff Bi-conic Capsule

Conf. 7

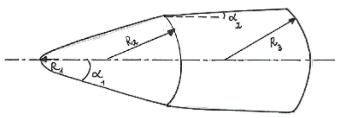


L/D = 0.55 – 0.7

The bluff bi-conic capsule is a shape envisaged by some German studies, in the frame of the European Crew Rescue Vehicle project.

Slender Bi-conic Capsule

Conf. 8

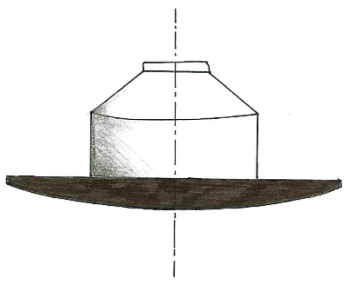


L/D = 0.8 – 1.2

The slender bi-conic capsule has been envisaged by Russians to carry out re-entry missions from Low Earth Orbits. In addition, some Mars Lander concepts have exploited this shape.

Heatshield with Afterbody

Conf. 9



L/D = 0.2 – 0.4

This configuration allows to guarantee the capability of withstanding extreme thermal loads to non-non winged configuration. With respect to the other proposed configurations, the heatshield is not part of the main body but it is a sort of external element. Depending on the specific application, it is also possible to envisage a detachable heatshield to be operated in specific mission phases. An example of implementation of this configuration is the Viking Mars probe.

Configuration	L/D
Sphere	0
Capsule	0.2 ÷ 0.4
Biconic	0.5 ÷ 0.7
Lifting Body	1.0 ÷ 1.4
Winged Body	1.4 ÷ 1.8

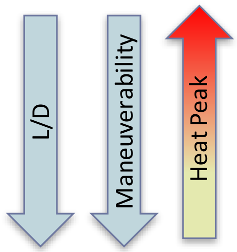


Figure 98: Different L/D configurations

As far as the aerodynamic configuration is concerned, the selection of the best alternative is strictly related to the stakeholders’ needs but can also be influenced by different constraints imposed by the trajectory, especially in case of mission including an orbital re-entry phase. As it has been done for the staging strategy, a high level trade-off has been carried out but in this case it is clear since the beginning that the only two configurations that are suitable for manned hypersonic transportation system are the lifting or the winged body configurations.

In order to guide engineers through the selection of the most suitable aero-thermodynamic configuration, an approach very similar to the one suggested and applied for the selection of the propulsive alternative and the take-off and landing strategy is presented. In particular, starting from the analysis of the stakeholders, the following list of requirements have been considered to have a certain impact on the selection of the take-off strategy to be adopted.

- The transportation system shall guarantee a proper accommodation to passengers
- The transportation system shall guarantee a proper accommodation to the payload
- The volumetric efficiency shall be maximized
- The transportation system shall maximize the lifting coefficient
- The transportation system shall guarantee proper stability and control characteristics during the overall mission phases
- The transportation system shall guarantee proper gliding performances

- Range in gliding
- Thermal loads shall be minimized
- Reusability of the entire transportation system shall be targeted.
- Maintenance actions shall be minimized
- Turn-around time shall be minimized
- The vehicle shall be hosted within conventional expendable launcher fairing.
- The capability of performing an autonomous take-off shall be guaranteed.
- The capability of performing soft landing shall be guaranteed.

Then, Table 41 summarizes the major areas of interest for this trade-off and for each of them, a list of drivers is presented.

Table 41: Areas of interest impacting on the selection of the best aerothermodynamic configuration

Areas of Interest for	**Drivers**	**Impact on aero-thermodynamic configuration**
Accommodation	• Comfort level	The required level of comfort is strictly related not only to the need of hosting humans on-board, but also to their level of physical preparation. Indeed, different levels of comfort can be required depending on the need of hosting astronauts or common not-trained passengers.
	• Volume efficiency	Different configurations can have a

		different volume efficiency.
Aerodynamic and Aero-thermodynamic	• Lifting coefficient	Each configuration is characterized by its proper aerodynamic characteristics.
	• Gliding performances	Depending on the aerodynamic characteristics of the selected aerothermodynamics configuration, the gliding performance can be changed.
	• Thermal loads	Thermal loads and heat peak shall be a direct consequence of the aero-thermodynamic configuration and the envisaged trajectory.
Stability and Control	• Stability	The selected configuration has a great impact on stability performances. In particular, different behaviors can been envisaged during the re-entry phase.
	• Maneuverability	Maneuverability is strictly related to the configuration through the possibility of hosting proper control surfaces.
Operations	• Reusability of the entire vehicle	In case the system shall be completely re-usable, the heat loads expected per mission shall be minimized.
	• Storable within launcher fairing	In case of non-autonomous vehicles, there will be the need of launching it through a conventional launcher. In this case, specific geometrical limitation will be imposed to the configuration.
	• Autonomous take-off	The capability of performing automatic take-off implies the need of hosting a proper propulsion system. In addition, capability of performing atmospheric climb shall be guaranteed.

	• Precise and Soft landing	The possibility of performing a precise and soft landing is directly related to the possibility of guaranteeing a proper level of controllability to the vehicle

It is worth noting that both maintenance and safety related issues are not currently considered as drivers for the trade-off but they are peculiar characteristics of each single configuration that can be investigated in strict relationship with the selected trajectory and subsystem level decisions. In addition, all the drivers have been evaluated for the different configurations presented in Table 42.

Table 42: Scoring strategy for the evaluation of the best aerothermodynamic configuration for a Re-entry Vehicle

		***Re-entry Vehicles* Configurations**								
	Drivers	**C1**	**C2**	**C3**	**C4**	**C5**	**C6**	**C7**	**C8**	**C9**
		Flying Wing	*Fuselage + Wing*	*Lifting Body*	*Spherical Capsule*	*Blunt Cone Capsule*	*Conic Capsule*	*Bluff bi-conic capsule*	*Slender bi-conic capsule*	*Heatshield with afterbody*
Accommodation	Comfort level	H(10)	H(10)	M(7)	L(3)	L(3)	L(3)	L(3)	L(3)	L(3)
	Volume efficiency	L(3)	L(3)	M(7)	M(7)	H(10)	H(10)	H(10)	H(10)	M(7)

Aerodynamic and Aero-thermodynamic	Lifting coefficient	H(10)	H(10)	M(7)	L(3)	L(3)	L(3)	L(3)	L(3)	L(3)
	Gliding performances	H(10)	H(10)	M(7)	L(3)	L(3)	L(3)	L(3)	L(3)	L(3)
	Thermal loads	L(3)	L(3)	M(7)	H(10)	H(10)	H(10)	H(10)	M(7)	H(10)
Stability and Control	Stability	H(10)	H(10)	M(7)	L(3)	M(7)	M(7)	M(7)	M(7)	M(7)
	Maneuverability	H(10)	H(10)	M(7)	L(3)	L(3)	L(3)	L(3)	M(7)	L(3)
Operations	Reusability of the entire vehicle	H(10)	H(10)	M(7)	L(3)	L(3)	L(3)	L(3)	L(3)	L(3)
	Storable within launcher fairing	L(3)	L(3)	M(7)	H(10)	H(10)	H(10)	H(10)	H(10)	H(10)
	Autonomous take-off	H(10)	H(10)	M(7)	L(3)	L(3)	L(3)	L(3)	L(3)	L(3)
	Precise and Soft landing	H(10)	H(10)	M(7)	L(3)	L(3)	L(3)	L(3)	L(3)	L(3)

Table 43: Sensitivity Analysis for the evaluation of the best aerothermodynamic configuration for a Re-entry Vehicle

Weighting Factor				**C1**	**C2**	**C3**	**C4**	**C5**	**C6**	**C7**	**C8**	**C9**
K_A	**K_B**	**K_C**	**KD**	*Flying Wing*	*Fuselage + Wing*	*Lifting Body*	*Spherical Capsule*	*Blunt Cone Capsule*	*Conic Capsule*	*Bluff bi-conic capsule*	*Slender bi-conic capsule*	*Heatshield with afterbody*
1/4	*1/4*	*1/4*	*1/4*	*1,58*	*1,58*	*1,23*	*0,63*	*0,80*	*0,80*	*0,80*	*0,98*	*0,80*
1/2	*1/6*	*1/6*	*1/6*	*1,48*	*1,48*	*1,28*	*0,75*	*0,97*	*0,97*	*0,97*	*1,08*	*0,97*
1/6	*1/2*	*1/6*	*1/6*	*1,62*	*1,62*	*1,05*	*0,28*	*0,40*	*0,40*	*0,40*	*0,62*	*0,40*
1/6	*1/6*	*1/2*	*1/6*	*1,72*	*1,72*	*1,28*	*0,62*	*0,87*	*0,87*	*0,87*	*1,12*	*0,87*
1/6	*1/6*	*1/6*	*1/2*	*1,48*	*1,48*	*1,28*	*0,85*	*0,97*	*0,97*	*0,97*	*1,08*	*0,97*

Considering the results of the trade-off analysis, it is clear that from a technical perspective, the exploitation of a more traditional transportation system configuration appears to be the most promising. However, this trade-off estimation does not take into account maintenance, safety and costs. They are strictly related to the specific case-study and for this reason they are not integrated here but an example of their impact on the selection is provided later on in this Chapter, for the suborbital vehicle selected as case study.

Besides the sensitivity analysis carried out giving different importance to the different drivers, the best configuration alternative result to be a flying wing aircraft or a more traditional fuselage plus wing configuration, immediately followed by the lifting body configuration. Looking beyond these conventional alternatives, the slender bi-conic capsule results to be the best one.

4.2 High Level Estimations

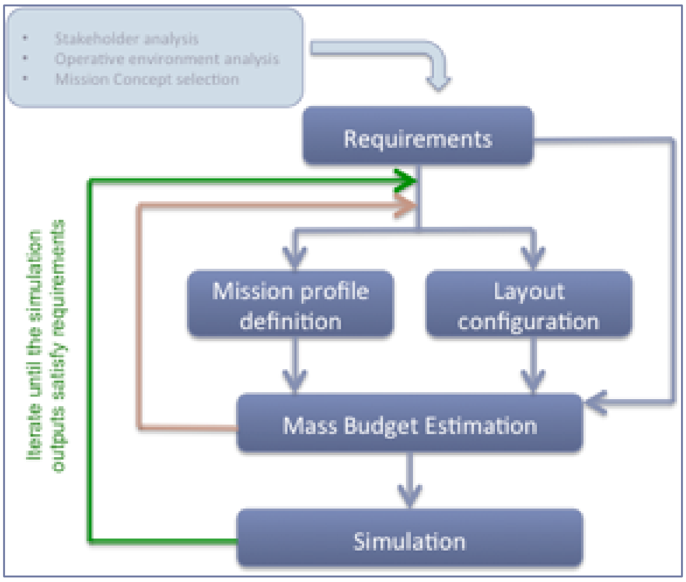


Figure 99: Role of High-Level Estimations in conceptual design

At this point of the conceptual design stage, phase, the configuration in terms of staging strategy, propulsive strategy and take-off strategy should have been fixed, together with the selection of the aero-thermodynamic configuration. Moreover, a vague idea of the transportation system layout should have been arisen, so that the first high level numerical estimations can be carried out. This is one of the most difficult activities, especially in the case of highly innovative transportation systems, where statistical analysis could be hardly carried out. However, the approach here suggested to obtain the very first numerical evaluations, makes benefit of the very few reference vehicles for which data are available. The process described in this paragraph, in order to comply with a MBSE approach have been reproduced in a Matlab® environment, with a proper GUI, and it has also been proved to be very useful in supporting the M.Sc. students of Integrated Systems Design classes. As it is sketched in the following Figure, the design process should start with the identification of the high level requirements that should affect these first numerical evaluations. In particular, they are strictly related to the payload type and estimate mass, the type of mission profile that has been envisaged, the layout and the very first choices in terms of staging propulsive and take-off and landing strategy. Please notice that the purpose of this preparatory activity is to select the proper criteria to filter the statistical population.

4.2.1 Mass Estimations

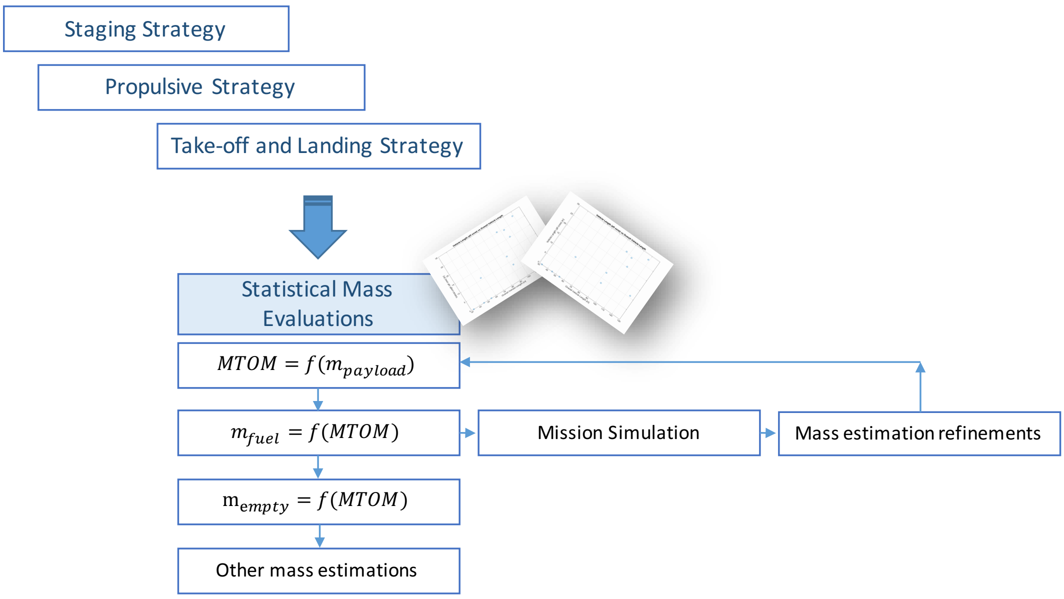


Figure 100: Flow chart summarizing the major steps of the very first mass estimations

The flow-chart reported in Figure 100 summarizes this high level estimation process. In particular, it is possible to notice that while the very first estimations are based on statistical data. They are mainly related to the high level mass estimations such as MTOM (Maximum Take-Off Mass), Fuel Mass, Empty Mass, and the first allocation in terms of fuselage, wing, landing gear and on-board systems masses. All these mass estimations will be furtherly refined. However, for some of them, especially for the fuel mass estimation (with a direct impact on both the MTOM and the empty mass), the exploitation of mission simulation will allow iteratively moving towards the most realistic value. In the following Figures, an example of statistical data elaboration is proposed. Data used in this example refers to hypersonic vehicles reported in (Harloff, 1988).

4.2.1.1 Maximum Take-Off Mass Estimation

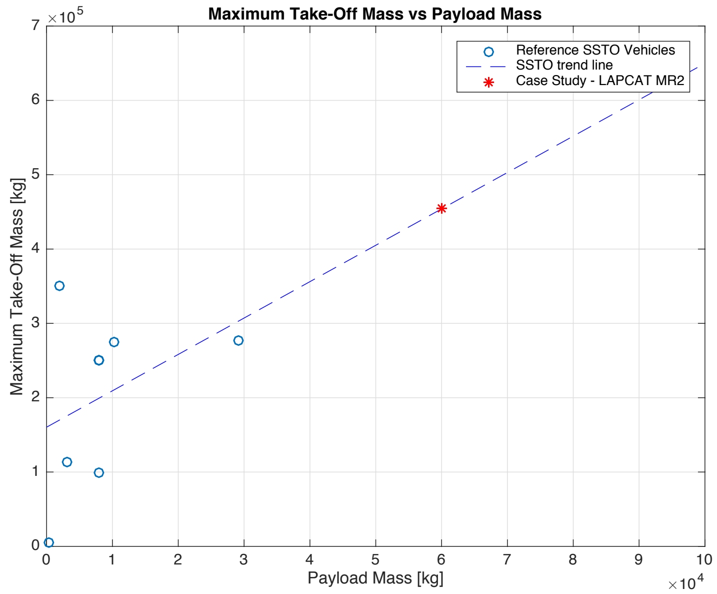


Figure 101: Maximum Take-Off Mass versus Payload Mass (Point-to-Point vehicles)

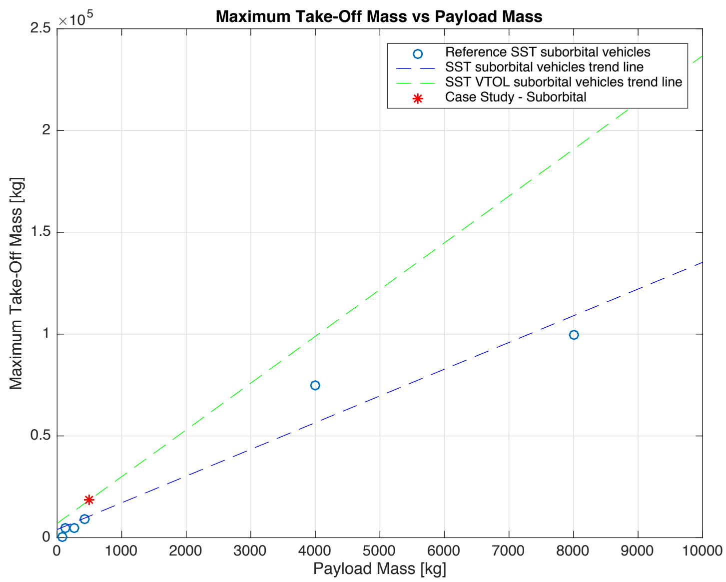


Figure 102: Maximum Take-Off Mass versus Payload Mass (Suborbital vehicles)

Table 44: Coefficient for the Estimation of MTOM

Vehicle Type	a_0	a_1
Point-to-Point	4,895	160345
Suborbital Vehicle	$4{,}286 \cdot 10^{-6}$	$4{,}796 \cdot 10^{-2}$
VTOL Suborbital Vehicle	22,968	7052,92

The MTOM can be then derived as a function of payload mass following the linear regression

$$MTOM = a_0 + a_1 m_{payload}$$

where the formula coefficient varies depending on the type of vehicle under investigation. As a result of a high level statistical analysis carried out with the available data of existing or under development concepts, the polynomial coefficient in Table 44 are suggested.

4.2.1.2 Propellant Mass Estimation

The propellant mass estimation is one of the most peculiar aspects to be specially considered for hypersonic vehicles and an estimation with acceptable confidence level can be reached through the exploitation of several iterations of mission simulations. However, it is important to succeed in reaching a good first estimation in order to initialize the mission simulation. To this purpose, in this context, the model proposed by HASA, the Hypersonic Aerospace Sizing Analysis for the Preliminary Design of Aerospace Vehicles (Harloff, 1988), has been considered.

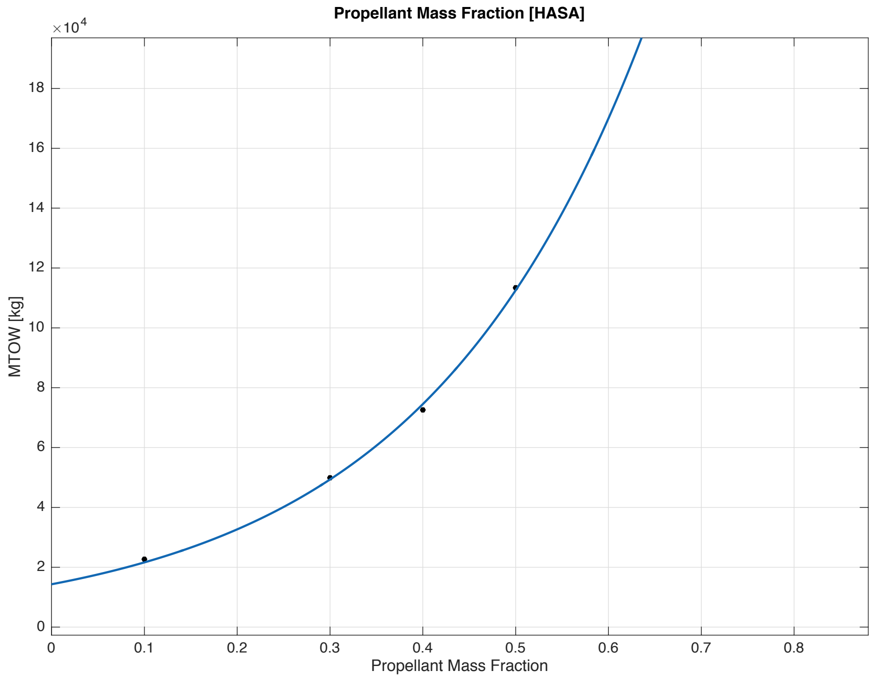


Figure 103: Maximum Take-Off Mass versus Propellant Mass Fraction

It highlights the dependency of propellant mass from both mission profile and vehicle configuration. Exploiting data coming from the reference document, it is an exponential form guarantees an optimal fitting curve. In particular, the MTOW can be expressed as:

$$MTOW = f(PMF) = k \cdot a\, e^{b \cdot (PMF)}$$

where,

$$a = 5{,}47 \cdot 10^4;\ b = 0{,}7042 \cdot$$

and PMF is the Propellant Mass Fraction.

Considering the fact that in case of a VTOL vehicle, a corrective factor (*k*) should be considered. Sensitivity analyses suggested that a factor of 1.17 is eligible for this kind of applications.

4.2.1.3 Vehicle length and diameter estimation

The Overall Vehicle Length is a variable and could not be estimated on the basis of the number of passengers only. A first sizing attempt will be carried out in Chapter 6, dealing with fuselage design. Indeed, as it will be detailed in Chapter 6, the overall vehicle length is mainly affected by the fuselage length that can be estimated since the conceptual design phase considering the need of accommodating crew, passengers and payload and integrating the required sub-systems. However, exploiting some statistical data, it is possible to have at least a rough idea of the length of the three major parts in which a vehicle can be divided longitudinally, with respect to the overall vehicle length:

- Forward cone ($L_{forward\ cone}$)
- Main body cone ($L_{main\ body\ cone}$)
- Aft cone ($L_{aft\ cone}$).

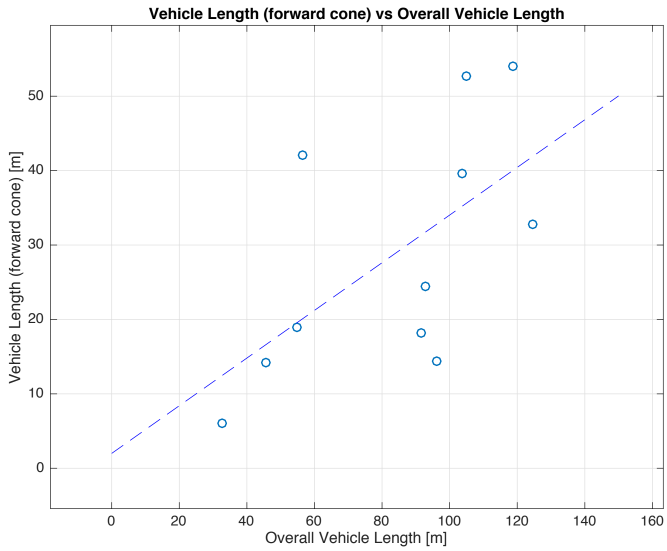


Figure 104: Vehicle Length (Forward Cone) vs Overall Vehicle Length

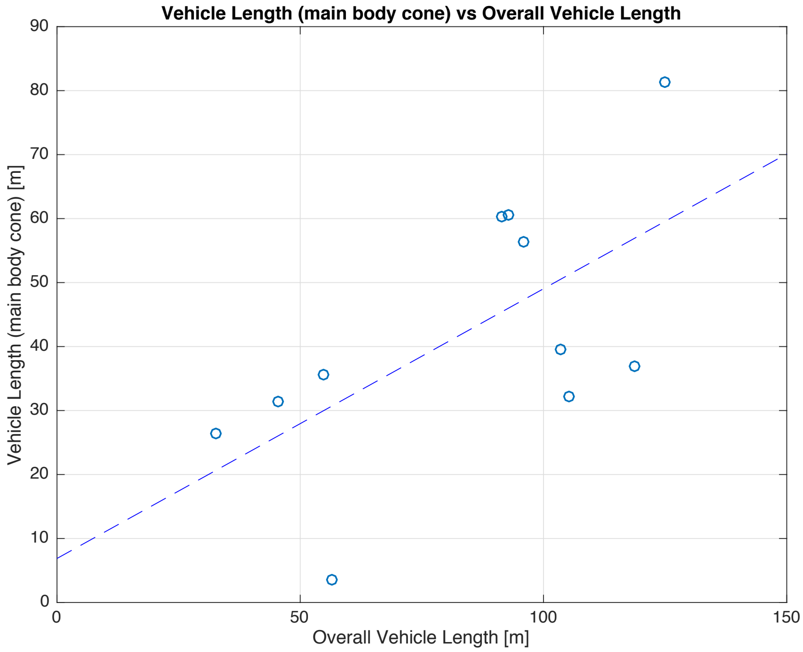


Figure 105: Vehicle Length (Main Body) vs Overall Vehicle Length

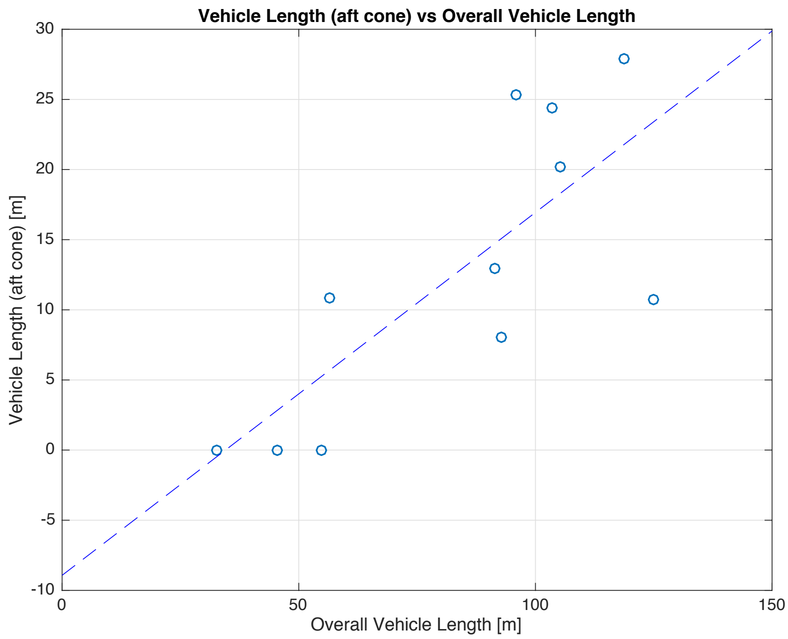


Figure 106: Vehicle Length (Aft Cone) vs Overall Vehicle Length

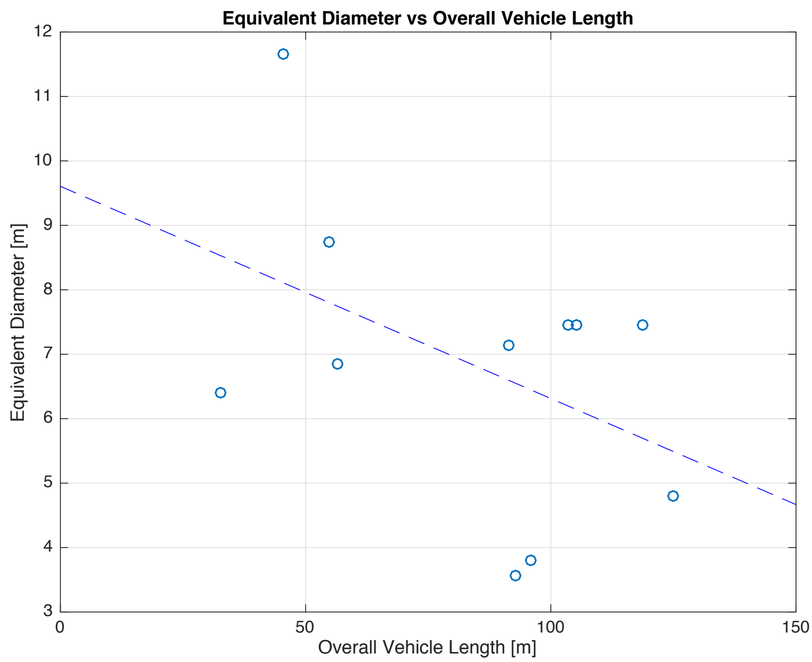


Figure 107: Equivalent Diameter vs Overall Vehicle Length

Following the same approach, the equivalent diameter can be estimated as function of the overall vehicle length.

$$L_{forward\ cone} = 0{,}320 \cdot L_{vehicle} + 1{,}997$$

$$L_{main\ body\ cone} = 0{,}421 \cdot L_{vehicle} + 6{,}874$$

$$L_{aft\ cone} = 0{,}259 \cdot L_{vehicle} - 8{,}926$$

$$D_{vehicle} = -0{,}0329 \cdot L_{vehicle} + 9{,}605$$

4.2.1.4 Wing Surface Estimation

As far as the wing surface is concerned, it should be estimated in statistical way, considering Wing Loading (W/S) and the MTOW.

Noticing from the trend line that:

$$\frac{MTOW}{S} = 0{,}0217 \cdot MTOW + 99{,}392$$

a first estimation of Wing Surface can be obtained.

This is another estimation that should be properly validated through the exploitation of mission and flight simulations. In particular, due to the need of generating lift at very low speed and angle of attack, proper values should of wing area should be considered for VTOL/STOL spaceplanes.

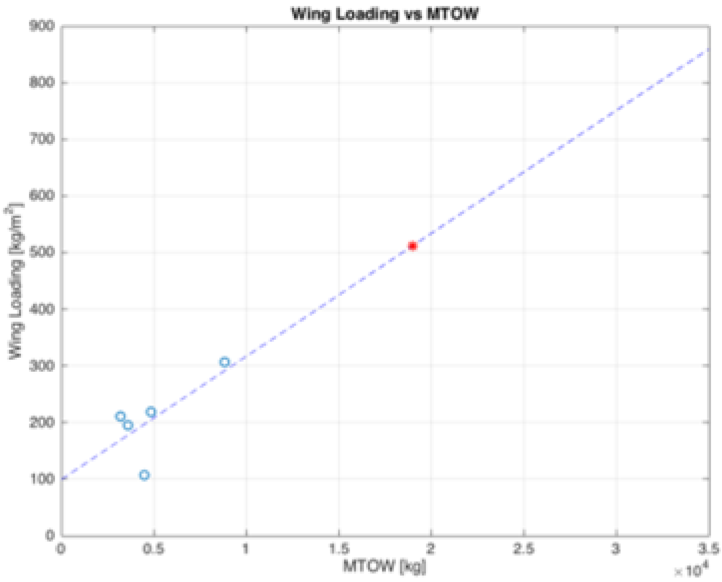


Figure 108: Wing Loading vs MTOM

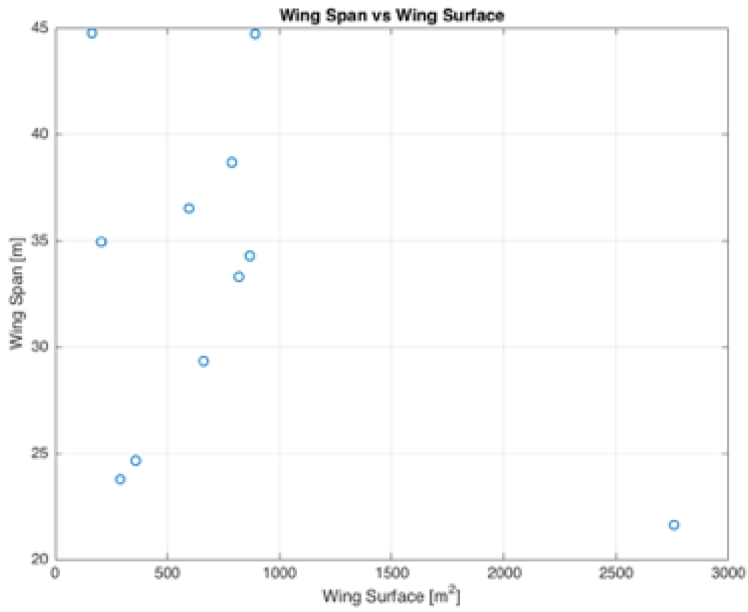


Figure 109: Wing Span vs Wing Surface

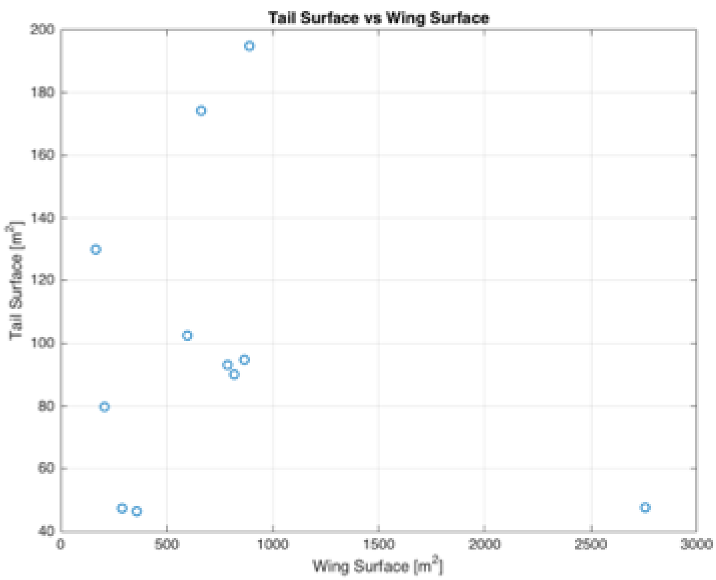


Figure 110: Tail Surface vs Wing Surface

Figure 111: Ad-hoc developed GUI to guide the designer through the statistical analysis, easing the iterations

It has to be noticed that in the same way, other characteristics such as the Maximum available thrust could also be estimated. Unfortunately, in some cases, like for the propellant mass estimation, these first values may be quite far from the final one and only simulations can be used to obtain numerical estimations with acceptable confidence level.

4.2.2 First geometrical sizing attempt

In addition to the suggested mass estimations, exploiting the statistical approach, it is also possible to have a first idea of the main aircraft dimensions, including tail surfaces. As it is possible to notice, the 2D or in some case 3D graphical approach is one of the most useful and widely exploited tool to represent statistical data. However, this is not the only way of working. There are also other ways to exploit statistical data, like for example to have an idea of the numerical ranges to be used. Some clear examples are reported in Table 45. They are always referring to (Harloff, 1988).

Table 45: Wing Geometrical Characteristics (high level statistical analysis)

Characteristic	Min	Mean	Max
Aspect Ratio	1.357	3.17	12.12
Wing Loading [kg/m^2]	251.937	485.58	711.04
Taper ratio	0.0960	0.2424	0.8
t/c	0.025	0.065	0.21

4.2.3 Mission Profile (First Sketch)

In addition to the identification of the most suitable criteria to filter out statistical data, there is also the need of generating a first draft of mission profile. At this stage, per each mission phase, the indication of the following characteristics is sufficient for the targeted purposes:

- Propulsive strategy (air-breathing mode, rocket mode, gliding mode etc.)
- Thrust level required;
- Time duration;
- Specific Fuel Consumption/Specific Impulse
- Starting and ending altitude

These data are mainly exploited within the process in order to obtain a second estimation of the overall fuel mass required to perform the mission. It is worth noticing that in this case, the mission profile is only presented in a descriptive for and no simulations hide behind these evaluations. An example of the high-level mission profile definition supported by the exploitation of an ad-hoc Matlab® tool.

Following the suggestions that could be directly gathered from the preliminary statistical estimations, it is possible to attempt a first mass allocation activity to the different design areas, as illustrated in Figure 113. Moreover, it is also possible to obtain a first level estimation of the aerodynamic characteristics of the vehicle, graphically estimating the lift and drag coefficients for different

Mach numbers and on the basis of L/D parameter. Indeed, the designer, with a general idea of the vehicle configuration, may select a proper reference vehicle and trying to extrapolate and then exploit the coefficients curves.

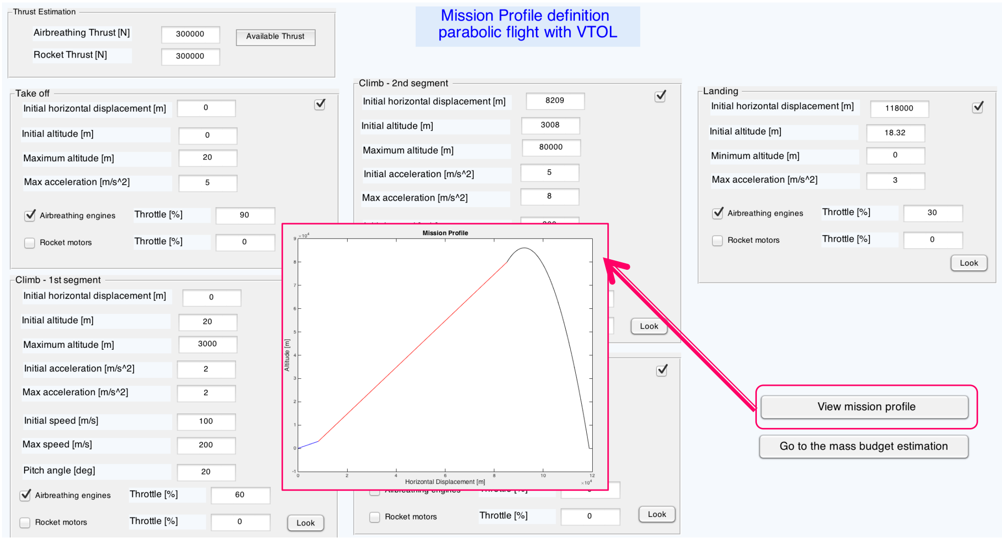


Figure 112: Ad-hoc developed GUI for the definition of the first mission profile parameters

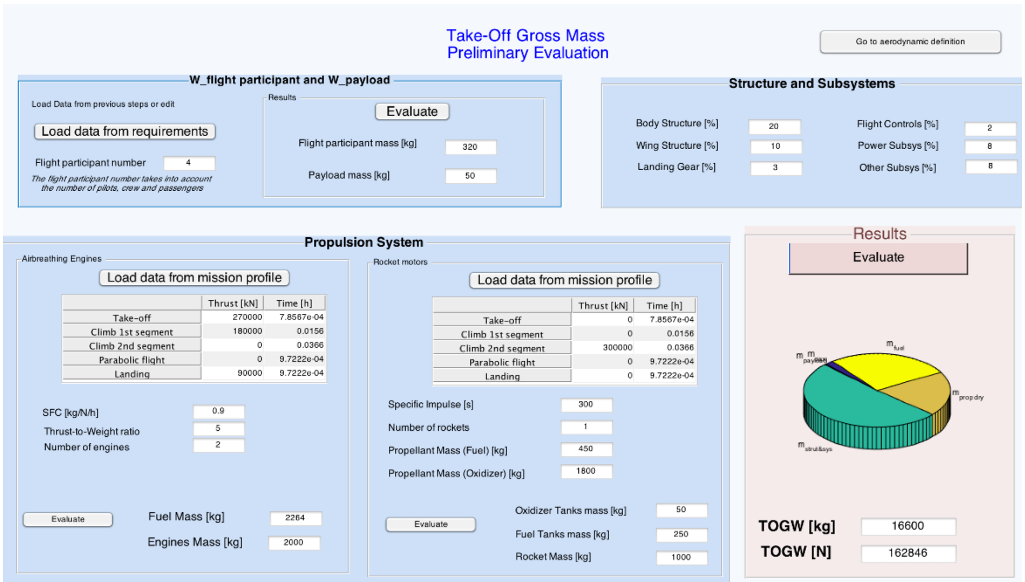


Figure 113: Ad-hoc developed GUI for the mass breakdown estimation

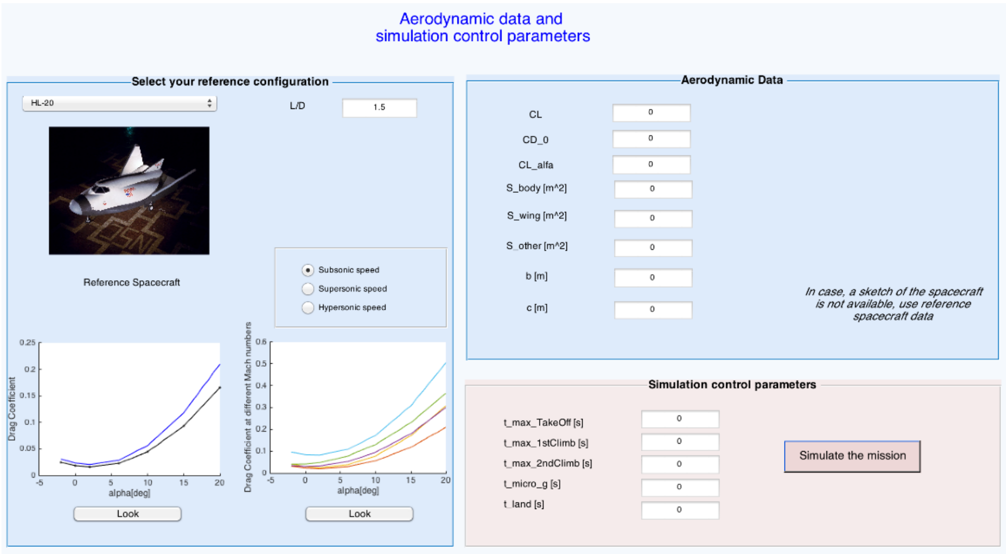


Figure 114: Ad-hoc developed GUI for the high level aerodynamic characteristics and simulation control parameters

Depending on the complexity of the system and related trajectory, mission simulation may be required since the conceptual design phase. In the application reported in the next section, two different mission simulation approaches have been used and compared. From one side, an ad hoc built-in tool, developed in Matlab® environment able to predict the mission profile of a suborbital vehicle have been properly developed. Additional details about this tool are reported in the next paragraph, together with its exploitation for the case study. The results obtained thanks to this tool have than been compared with the outcomes of a much more complex and high-level precision commercial software. ASTOS® has been selected among all the available software because, thanks to its flexibility, it can also be used during the very first sizing attempt. Indeed, it is not forcing the user in inserting too many specific inputs that can be unknown at this design stages. In any case, the evaluation of the fuel mass to be stored on board the vehicle appears to be a highly iterative and recursive process. Whatever kind of support tool is selected and used, the value of fuel mass obtained from the statistical analysis it is inserted to describe the initializing condition. Starting from this point, the mission simulation is performed and the relative results should be properly analysed considering some specific targets. As far as trans-atmospheric vehicles are concerned, the maximum altitude to be reached can be used as control parameter for a suborbital mission profile, while the maximum range or endurance seems to be most suitable for a point-to-point mission. As it sketched in Figure 115 it is possible to see not only the importance of understanding the impact of the several design input variables on the simulation results but also understanding the mutual relationships existing between the same input variables.

For example, in the case of a suborbital mission profile, three different design parameters can be modified in order to reach the envisaged target altitude. Indeed, playing with the available propellant mass, the available mass and the specific impulse, the mission profile can be heavily modified. Unfortunately, it is very difficult to establish a direct correlation between input modifications and expected output because of the high number of interconnections among variables. For example, some variations of the propellant mass imply a TOGW variation not only due to the higher propellant mass fraction, but also to the higher dry mass because of the increase in tanks volume.

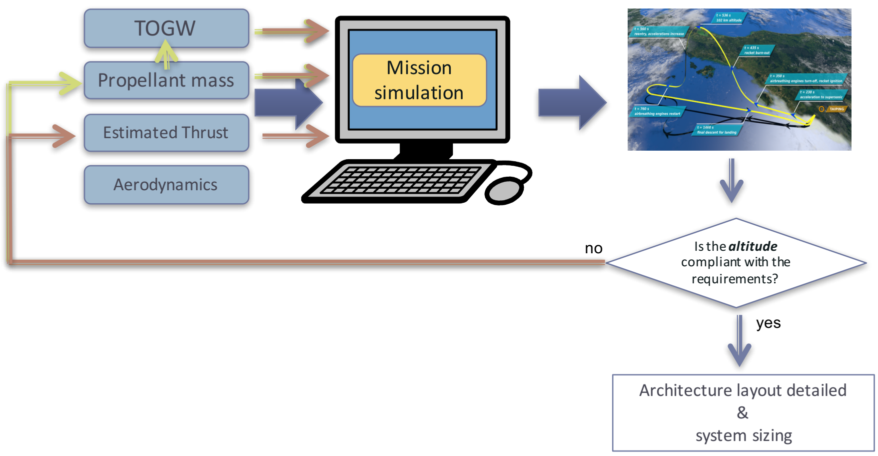


Figure 115: Role of mission simulation within the iterative and recursive conceptual design stage

4.3 Case Study: System architecture alternative and high level estimations for a suborbital vehicle.

In this section, the selection of the best alternative in terms of staging strategy, propulsive strategy, take-off and landing strategy and aerothermodynamic configuration are reported for the specific case-study, following the theoretical process proposed above. In addition, the first high level estimations for the selected architecture are performed and reported.

4.3.1 Staging Strategy selection

Staging strategy has been investigated following the trade-off analysis approach described at the beginning of this chapter. Considering the fact that, looking at the need expressed by the stakeholders, safety should be the most impacting Figure of Merit, the results of the trade-off analysis identify the best alternatives as:

- A two-stages transportation system exploiting an already existing carrier.
- Single stage suborbital vehicle.

After a new cycle of revision with the stakeholders, the two-stages alternative has been discarded for high level strategic decision. First of all, the Malaysian stakeholders, major players of this enterprise, would like to promote their own industrial capabilities, without the need of buying existing manufacturing products (e.g. the carrier) from abroad. In addition, in order to ease the operations and to target a future routine service, the Single Stage alternative appears to the stakeholder to be the best alternative.

Staging strategy: ***Single stage***

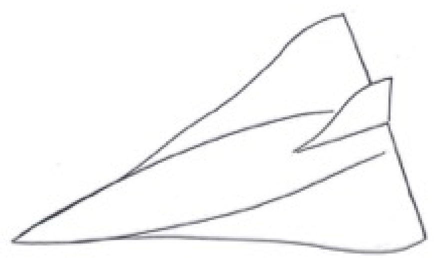


Figure 116: Selected staging strategy for the case study

4.3.2 Propulsive Strategy

Considering the propulsive strategy, additional constraints coming from existing regulations of the nation in which the vehicle will be exploited, prevent the designer from exploiting rocket engines up to a certain altitude. For this reason, looking at Section 4.1.3, the propulsive configurations 6 and 8 cannot be applied. Moreover, the additional stakeholder requirement of guaranteeing the shortest possible time-to-market, forces the engineers to envisage the usage of matured technologies. It is mainly for this reason that, besides the fact that from a theoretical perspective, Conf. 5 (Turbojet/Ramjet/Rocket) results to be the best one, Conf. 7 has been considered the best alternative to be implemented. Summarizing, the Single-Stage should be equipped with:

- an air-breathing propulsive sub-system able to support the vehicle when it is operated in lower atmospheric environment;
- a rocket-based propulsive sub-system able to support the vehicle during the ascent phase.

It is clear that the integration of these two subsystems within the same a single vehicle stage will be one of the major challenges from the systems integration perspective. Moreover, as it is explained in the following section and in-depth analysed later on in this thesis, the propulsive systems on-board integration will be deeply affected by the selected take-off and landing strategy too.

Propulsive strategy: TJ with Afterburner + Rocket

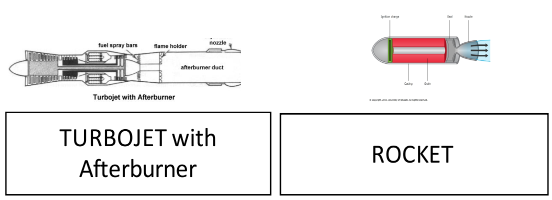


Figure 117: Selected propulsive strategy for the case study

4.3.3 Take-off and Landing Strategy

As far as the take-off and landing strategy is concerned, the trade-off proposed in the previous sections have been applied. It is possible to notice that considering the extreme importance of accommodation and aerodynamics and aero-thermodynamic issues among all the requirements, weighting factor selection played a fundamental role.

Both the selected weighting factors and the results of the trade-off analysis have been summarized in the following table. As it is possible to notice, the first ranked alternative is the configuration that exploited the diversion of the hot exhaust gases. However, this configuration may suffer from some problems during transition phase between the vertical and the horizontal flight. It is mainly for this reason that the addition of proper steerable nozzles will be considered. As far as safety is concerned, as reported in Chapter 7 dealing with integration issues, the number of nozzles, their modes of operation and their placements, have been furtherly analysed.

Table 46: Detailed of the scoring strategy for the selection of the optimal take-off and landing strategy for the reference suborbital vehicle

Weighting Factor						C1	C2	C3	C4	C5	C6	C7	C8	C9
K_A (Accommodation)	**K_B (Aerodynamic)**	**K_C (Structure)**	**K_D (Operations)**	**K_E(Control)**	**K_F (Logistic)**	*Tail Sitting*	*Vectored Trust at CG*	*Tilt Nacelle at CG*	*Un-augmented flow*	*Tip driven fan*	*Ejectors*	*Separate Lift Engines*	*L+L/C vectored*	*L+L/C tilt nacelles*
1/3	*1/3*	*1/12*	*1/12*	*1/12*	*1/12*	-0,4417	-0,3167	-0,5083	0,0333	-0,1667	-0,2667	0,1167	-0,3917	-0,7833

Take-off and Landing strategy:	*VTOL with exploiting diverted flow*

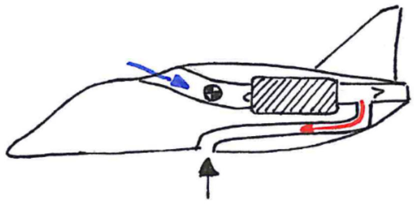


Figure 118: Selected take-off and landing strategy for the case study

It is worth noting that this kind of propulsion system architecture allows diminishing possible thermodynamic problems due to the extremely high temperature that the bottom side of the vehicle can experience during the re-entry phase. During this part of the trajectory, i.e. when the air-breathing subsystem is no more active, steerable nozzles should be retracted within the vehicle main body and structural interruptions required to host such nozzles in the extracted position should be properly sealed.

4.3.4 Aerothermodynamic configuration

Considering the aerothermodynamic configuration, the most suitable configuration that can allow to meet all the stakeholders' needs considers the fuselage/wing assembly. In fact, a more traditional configuration can allow hosting the envisaged propulsion subsystems and performing the vertical take-off and landing strategy.

Aerothermodynamic and aero-thermodynamic configuration:	*Fuselage + Wing configuration*

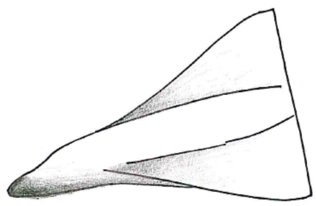


Figure 119: Selected aerothermodynamic configuration for the case study

It is important to notice that this is only a first hypothesis on the aerothermodynamic configuration class and the final vehicle layout may be very different from this first sketch. However, also in the final configuration, the two major elements, fuselage and wing, will be ease to be recognized.

4.3.5 High-Level estimations

The following tables summarizes some high level estimations both in terms of mass breakdown that in terms of generic geometrical characteristics for the selected case study.

Table 47: High level mass estimations for the reference suborbital vehicle

High Level mass estimations	Value	Reference
MTOM [kg]	1.854e+04	Assuming a payload mass of 500 kg
Propellant Mass Fraction Estimation	0,0632	See Figure 103
Fuel Mass [kg]	1998	See Figure 103 and taking into account a corrective factor of 1,17 (VTOL configuration)

4.3.6 Simulations

Simulations play a crucial role since they are expressly performed in order to retrieve useful data to verify the effectiveness of preliminary assumptions and then to propose, at each iteration, all the required adjustments. In doing so, there are more chances that system requirements are met in a shorter time. In particular, in order to enhance the confidence level in the statistical mass estimations, both an ad-hoc simulation code and a commercial mission simulation software have been exploited. In this way, the results obtained with the ad-hoc built in tool has been properly compared with the higher fidelity commercial software in order to validate the simplified model used at this design stage. The program proves to be useful also under various aspects. The simulations provide a general overview of the studied mission, giving context to the outputs and thus allowing to better understand them. They allow highlighting criticalities and the implications of newly introduced modifications, consequently permitting to underline, or find, the relationships between the characteristics of the system. Lastly, the program can assist in the search for derived characteristics and requirements, both of the system and of the support infrastructures, thus enriching the preliminary analyses with detailed information grouped and correlated with numerical and graphical data.

4.3.6.1 Ad-hoc built in mission simulation tool

The ad-hoc built-in simulation code was developed in order to determine the best mission profile for a given vehicle configuration (De Vita, 2015), (Fusaro 2017b). Moreover, the simulation code aims at optimizing (hence minimizing) important configuration and systems parameters like fuel consumption and maximum take-off weight of the vehicle. The vehicle and the environment in which it operates are modelled with a set of rigorous rules formally implemented in the form of mathematical equations. Therefore, the overall system being simulated is described by means of a set of proper mathematical models. In particular, the tool has been conceived with a modular architecture allowing enough flexibility to be exploited for a sufficiently high range of applications. In particular, it allows generating a mission profile putting together at least two of these mission phases:

- vertical take-off
- transition from vertical take-off to the horizontal flight
- initial atmospheric flight ascent
- transition from air-breathing engines to the rocket engine
- rocket engine burn-out and coasting
- descent, air-breathing engine restart and landing

In order to enhance the user interaction with the tool, the most interesting results of each simulation is presented both in numerical data from and graphical form. Moreover, the ad-hoc built-in tool has been conceived to be used both in a standalone mode, and within an integrated tool chain, with the possibility of automatically receiving inputs from other high level sizing tools, inserting simulation within the iteration loops aiming at giving to the users a final convergence value.

The under investigation suborbital mission requires the vehicle to climb far beyond the aeronautical heights and up to the altitude of 100 km or above. It means that the vehicle has to deal with different flight and atmospheric conditions and it is mandatory therefore to implement a proper atmospheric model. Actually, the implemented atmospheric model consists of three different sub-models, dealing with a particular atmospheric region each:

- the troposphere model, for altitudes comprised between the sea level and up to 11 km;
- the stratosphere model, for altitudes above 11km and up to 25km;
- a reduced version of the Jacchia Reference Atmosphere model for altitudes above 25km

Considering the special requirements of the case study, the mission starts with a vertical take-off. At this design stage, no assumptions have been done for the system that will allow to perform this mission phase. For the purposes of this high level design simulations, the implemented model describes a virtual steerable nozzle, directly acting in the aircraft aerodynamic centre and pointing towards the ground in order to overcome the weight of the vehicle with its thrust. Proper balancing actions should be envisaged, at the same time, in order to maintain the horizontal attitude of the vehicle during the first pure vertical transitional phase. When a predetermined altitude, imposed by safety regulations, has been reached, the thrust vector should rotate allowing the generation of a horizontal velocity. This phase is traditionally called as transition to the horizontal flight and it ends when the thrust vector is aligned to the vehicle longitudinal axis or when the desired angle of attack for the following mission phase is reached. With this model the code is also simulating, with a proper fidelity, what happens during the take-off using steerable nozzles underneath the vehicle to perform a non-tail sitting VTOL and then switching to the pure longitudinal thrust using the same engines (Figure 120).

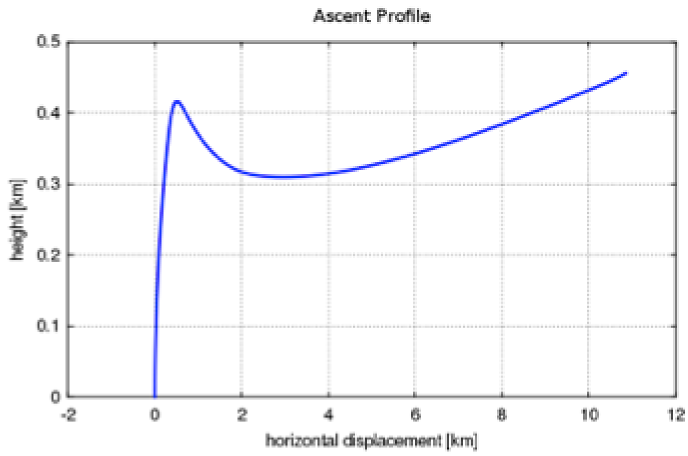


Figure 120: Example of mission simulation output (detail of the transition between vertical and horizontal flight)

Once the vehicle has reached the horizontal flight attitude, the first climb phase can be performed. In this specific case study, the ascent consists of several segments, depending on the adopted propulsive strategy. After the vehicle has reached a certain planned altitude and just below the service ceiling, the vehicle cannot use the air-breathing engines anymore and the transition from the air-breathing engines to the rocket engine takes place. In order to simplify the mathematical model, the rocket engine is supposed to be activated in the same moment in which the air-breathing engines are switched off. From this point, the vehicle completes the climb in an un-powered mode, until it reaches the target altitude of (more than) 100 km. Reaching this goal, the ascent phase is concluded.

To cope with the simulation of these mission phases, slightly different mathematical models in terms of attitude and thrust controls have been implemented. Notwithstanding, each model provides as output, forces and moments to ensure vehicle stability through the overall mission profile. A certain control over the thrust magnitude exists in order to prevent any possibilities that the vehicle overcomes the accelerations limits and, moreover, the flight trajectory and the thrust controls are optimized in order to make the amount of fuel to be consumed as low as possible.

Figure 121 outlines in a graphical way the main structure of the ad-hoc developed tool.

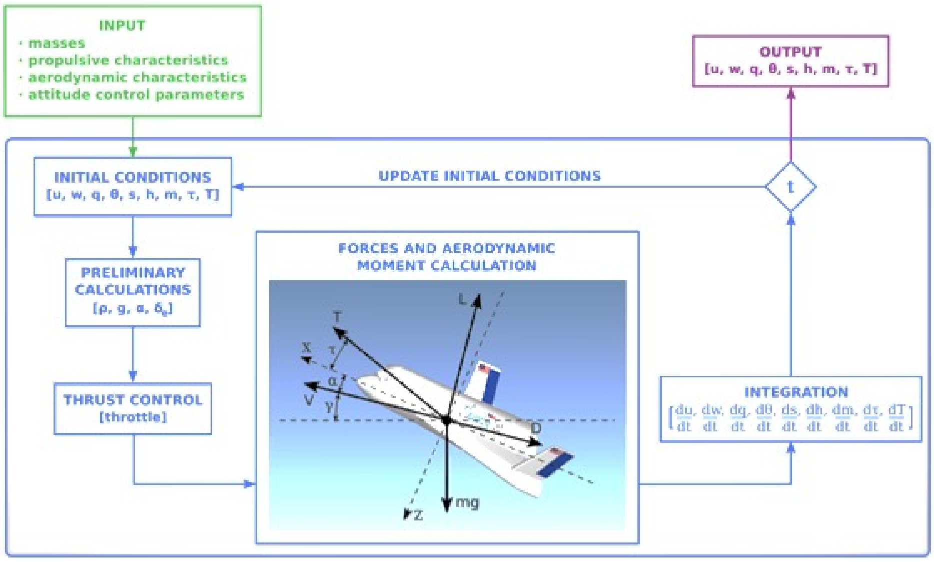


Figure 121: Structure of the ad-hoc developed mission simulation tool

For the sake of clarity, it is worth noticing that forces and moments are evaluated in the body axes longitudinal reference frame of the vehicle in which the X axis is parallel to the fuselage reference line, the Z axis is perpendicular to the X axis and it is directed from the upper to the lower surface of the wing airfoil. Moreover, from this scheme, it is possible to notice that the simulation is an iterative process, which updates the initial conditions by integrating a set of differential equations in the defined time frame. The initial conditions are evaluated on the basis of the input data and they are updated at each iteration thus defining the status of the vehicle in terms of velocity, attitude, position, weight and thrust at each moment of the simulation.

Looking at the major outputs of these simulations, it is possible to notice that the re-entry phase has been simulated. Indeed, the problem of re-entry of hypersonic vehicles it is too complex to be implemented in a simplified way within the conceptual design phases. However, a very simple routine able to simulate a ballistic re-entry has been implemented without any aero-thermo-dynamic corrective factors. Furthermore, in order to comply with additional stakeholder requirements, a “cruise-back” routine has been implemented, allowing the vehicle landing in the same place from which it took off. It is clear that this mission phase can be activated only in case a sufficient amount of fuel is still available on-board.

Exploitation of the ad-hoc built in tool within the tool-chain

The simulation tool described in the previous section can be used in a standalone mode or within the integrated conceptual design tool chain and exploited to refine the sizing of the different elements (Figure 122). The first sizing iteration has been initialized with the data obtained during the statistical estimation phase. In particular, the first mass estimation suggested a TOW of about 20000 kg. With these set of inputs, it is possible to notice (see Figure 123) that a maximum altitude of 65 km could be reached. In order to allow the vehicle to be compliant with the requirements stating that a maximum altitude of at least 100 km shall be reached, a series of iterations have been performed.

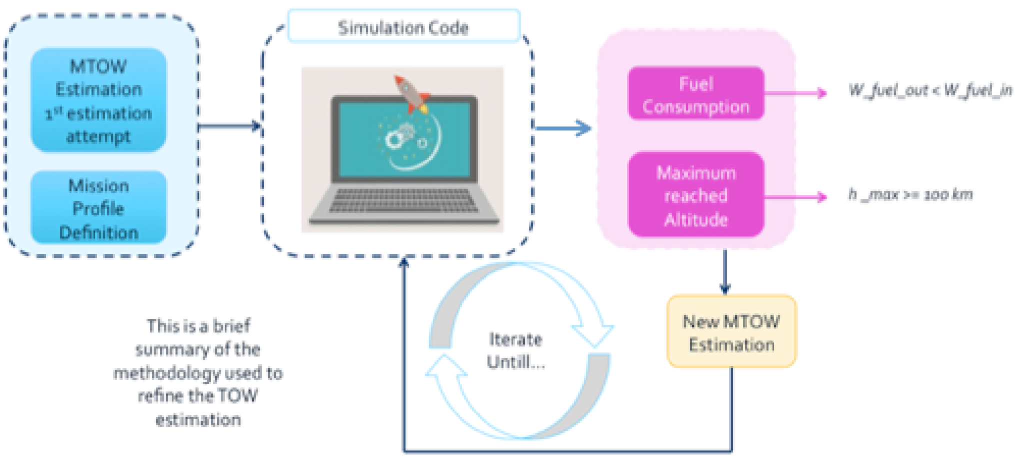


Figure 122: Iterations scheme for the mission simulation with the ad-hoc developed tool

			Input at simulation #1
INPUT	W_airbreathing	kg	7300
	W_fuel	kg	2500
	W_rocket	kg	200
	W_propellant	kg	4000
	W_propulsion_dry	kg	7500
	W_propulsion_wet	kg	14000
	W_strutt	kg	3000
	W_syst+payload	kg	3000
	MTOW	kg	20000
	T_rocket	kN	140
	T_airbreathing	kN	300
OUTPUT	W_fuel	kg	2000
	h_top	km	65

Figure 123: First iteration: inputs and outputs

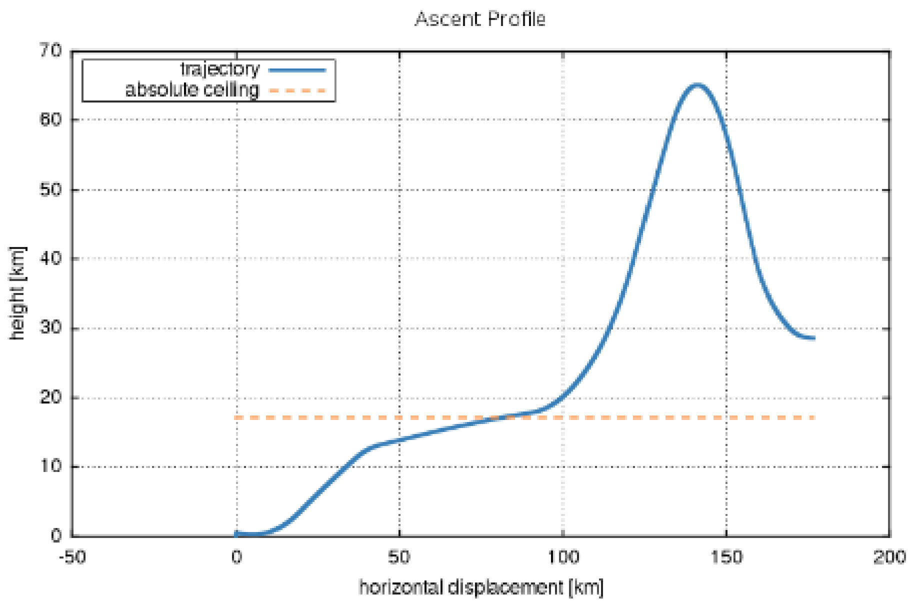


Figure 124: Example of graphical output of the ad-hoc developed tool after the first iteration.

After seven iterations, the simulation results indicate that the vehicle can reach the imposed target altitude with a MTOW of about 23300 kg.

			Input at simulation #1	Input at simulation #2	Input at simulation #3	Input at simulation #4	Input at simulation #5	Input at simulation #6	Input at simulation #7
INPUT	W_airbreathing	kg	7300	7300	7300	7300	7300	7300	7300
	W_fuel	kg	2500	2500	2500	2500	2500	3000	5000
	W_rocket	kg	200	220	303	364	364	364	364
	W_propellant	kg	4000	4000	4500	5000	5000	5000	5200
	W_propulsion_dry	kg	7500	7520	7603	7664	7664	7664	7664
	W_propulsion_wet	kg	14000	14020	14603	15164	15164	15664	17864
	W_strutt	kg	3000	3000	3000	3000	4000	4000	4000
	W_syst+payload	kg	3000	5000	5000	5000	3000	3000	1500
	MTOW	kg	20000	22020	22603	23164	22164	22664	23364
	T_rocket	kN	140	181	250	300	300	300	300
	T_airbreathing	kN	300	300	300	300	300	300	300
OUTPUT	W_fuel	kg	2000	2100	2150	2160	2193	2193	2193
	h_top	km	65	65,2	83	98	108	103	102

Figure 125: Summary of the results of the various simulation iterations.

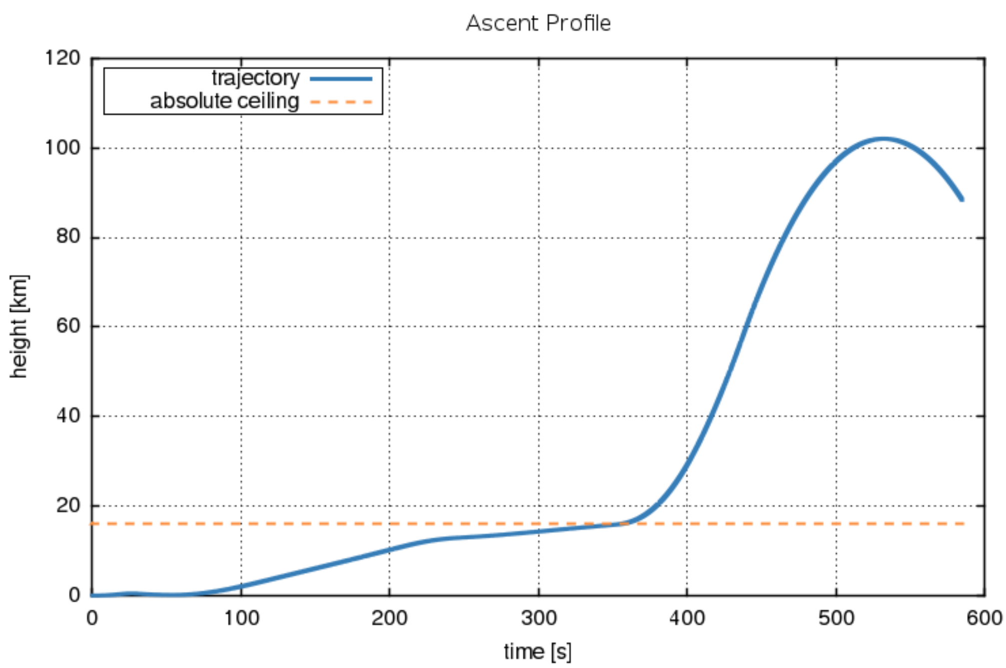


Figure 126: Example of graphical output of the ad-hoc developed tool after the last iteration (altitude vs time).

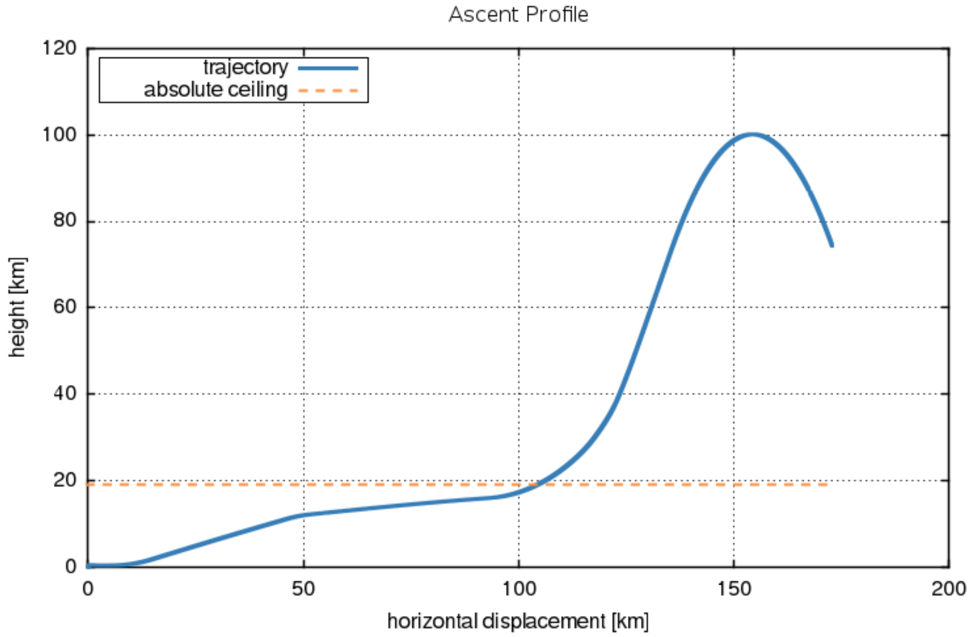


Figure 127: Example of graphical output of the ad-hoc developed tool after the last iteration (altitude vs horizontal displacement).

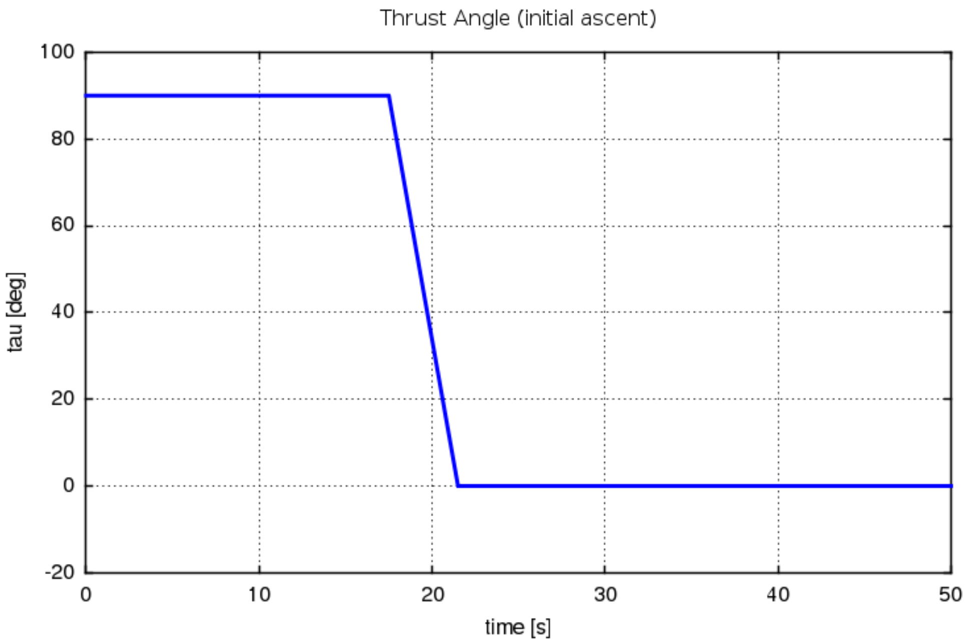


Figure 128: Example of graphical output of the ad-hoc developed tool after the last iteration (thrust angle vs time).

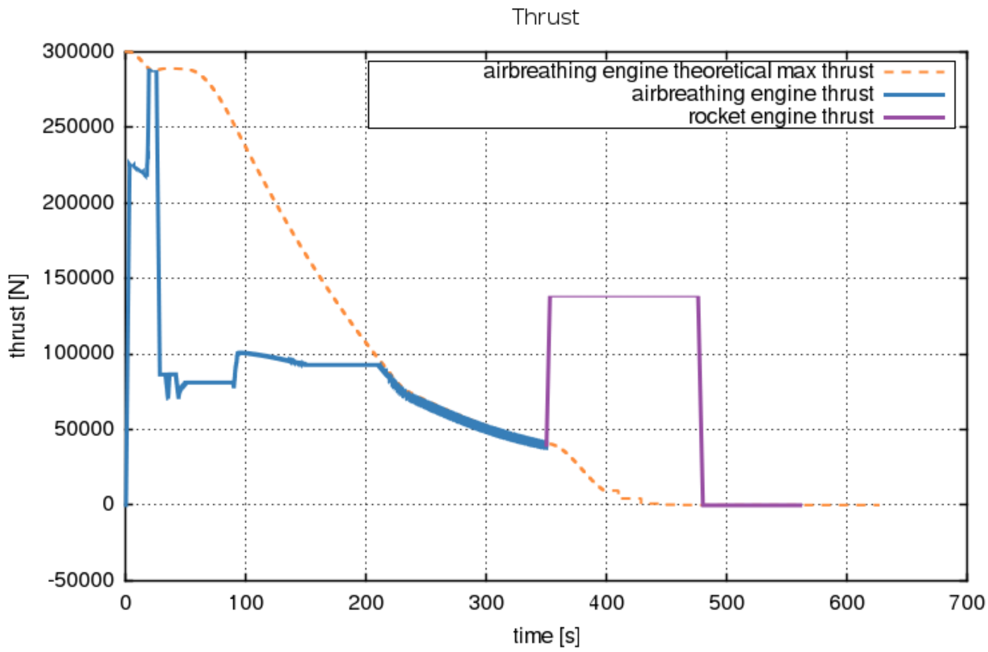


Figure 129: Example of graphical output of the ad-hoc developed tool after the last iteration (thrust vs time).

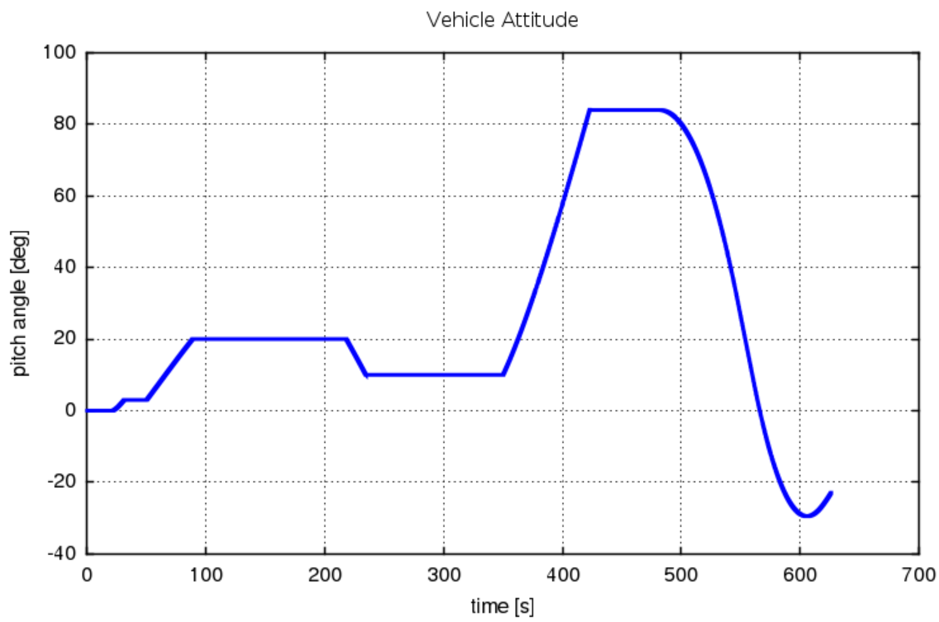


Figure 130: Example of graphical output of the ad-hoc developed tool after the last iteration (pitch angle vs time).

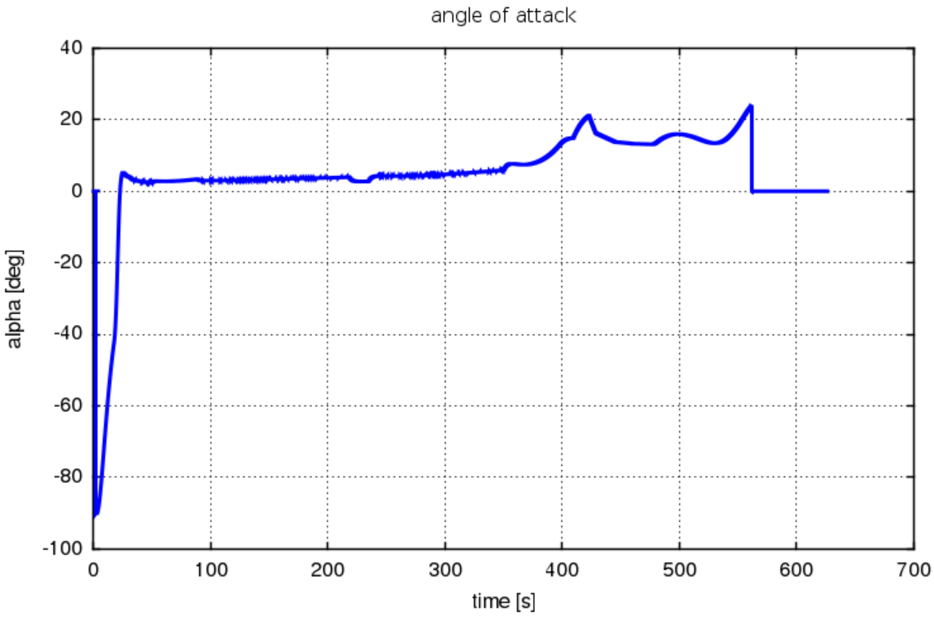


Figure 131: Example of graphical output of the ad-hoc developed tool after the last iteration (angle of attack vs time).

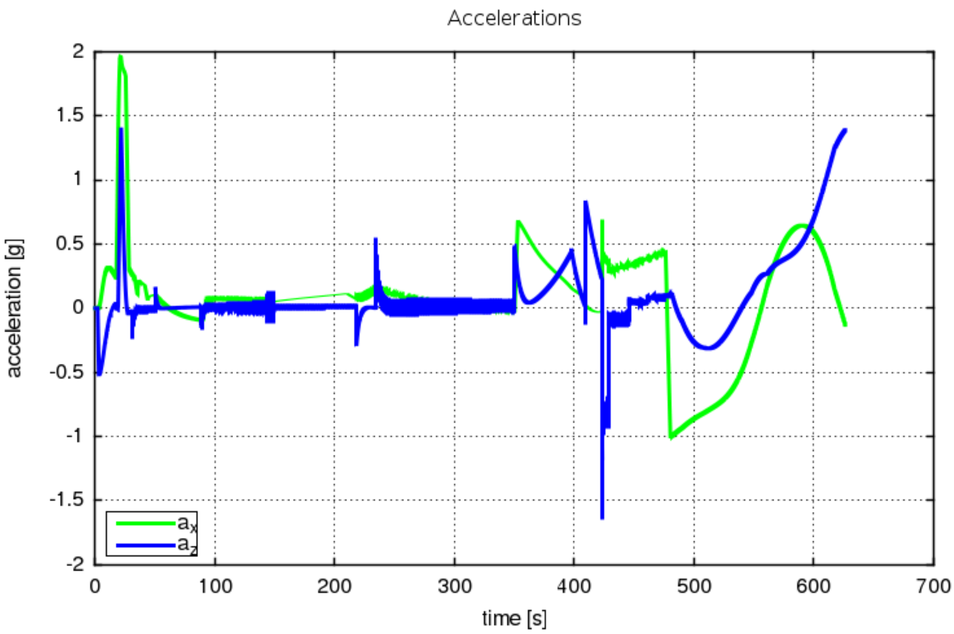


Figure 132: Example of graphical output of the ad-hoc developed tool after the last iteration (accelerations vs time).

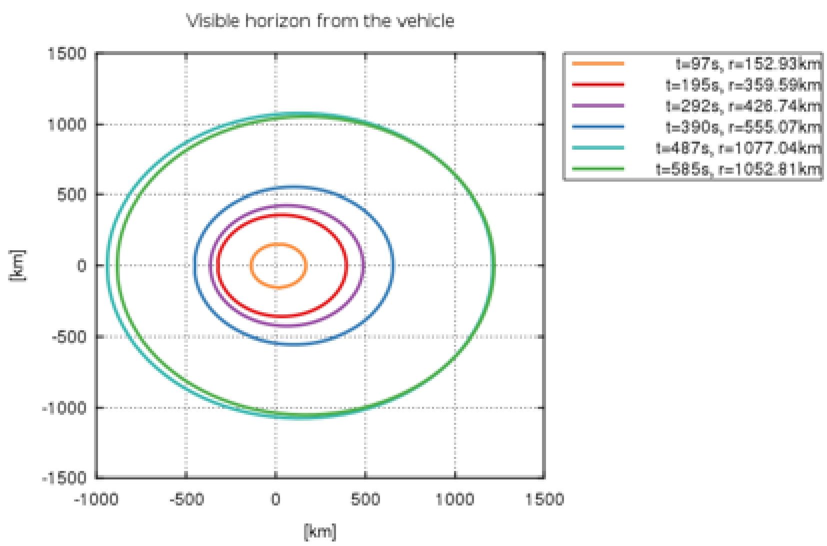


Figure 133: Example of graphical output of the ad-hoc developed tool after the last iteration (visible horizon from on-board at different simulation time).

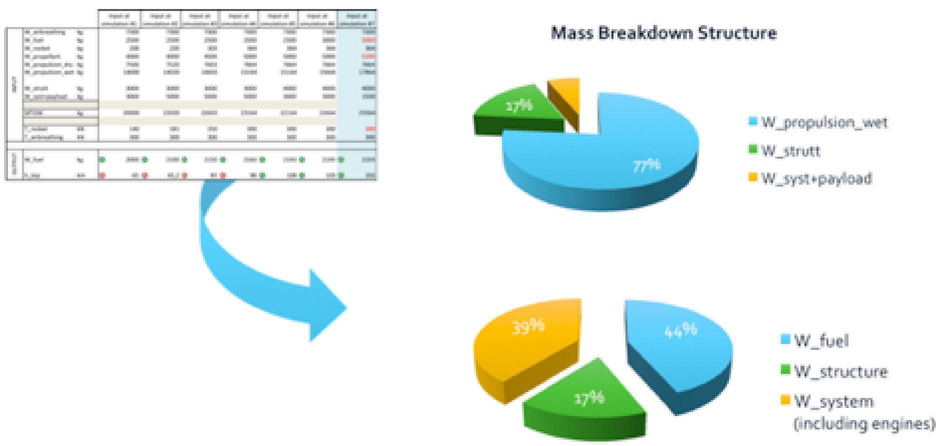


Figure 134: Mass Breakdown after the simulation iterations

Looking at the results obtained after seven iterations of the mission simulation exploiting the ad-hoc built-in tool it is possible noticing that the vehicle is able to fulfil the mission requirements whether equipped with sufficiently powerful propulsion subsystems and with an adequate amount of fuel and propellant. Besides the very limited number of inputs required by this mission simulation tool, the software allows accessing to a noticeable number of outputs that are very useful especially at this design stage. For example, the graphical output reporting the acceleration profile all along the mission is very important because it can be used to verify the compliance of safety requirements, mainly related to the fact that non-trained passengers shall be accommodated. In addition, these values can be considered as first estimations to initialize specific discipline detailed evaluations.

4.3.6.2 Mission Simulation with the commercial software ASTOS®

With the aim of both validating the results obtained with the mission simulation built-in tool and of providing a meaningful alternative to support the first high level estimations with mission simulation, the commercial software ASTOS was used. ASTOS, AeroSpace Trajectory Optimization Software, is an extensive user library of aerospace vehicle differential equations for 3-DOF and 6-DOF, aerospace vehicle boundary and path constraints as well as cost functions. Using the ASTOS library a multitude of launch- and re-entry vehicles and missions can be modelled solely by data input such that a user does not have to code anything. Advanced mission analysis is possible through the export filter to STK. As templates, several vehicles and missions are provided.

As far as suborbital mission profiles are concerned, ASTOS ARE® - Amateur Rocket Edition – a simplified version of the software, can be exploited.

In the following figures, some results obtained with the exploitation of ASTOS and the comparison with the previously gathered outcomes coming from the exploitation of the ad-hoc built-in tool are reported. In some cases, the major differences between these latest results and those obtained after the seventh simulation with the ad-hoc built-it tool are mainly related to different models laying behind the simulation, or to the difficulties in initializing such a complex mission simulator with data available at conceptual design stage.

It is also important noticing that the exploitation of a commercial software aimed at performing mission simulation, besides guaranteeing results with a higher confidence level, allowed to validate the model and the tool ad-hoc developed, paving the way for several other improvements.

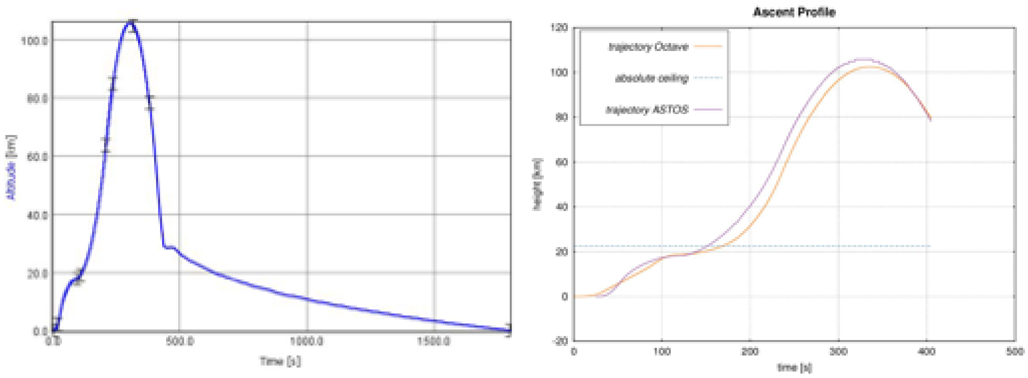


Figure 135: Mission profile with ASTOS (left) and comparison of the ascent phase with the results obtained with the ad-hoc built-in tool (right)

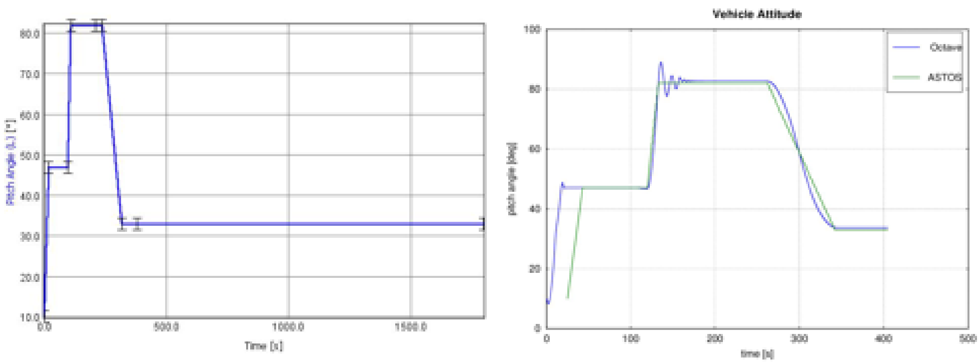


Figure 136: Pitch angle variation during the overall mission time with ASTOS (left) and comparison of the ascent phase with the results obtained with the ad-hoc built-in tool (right)

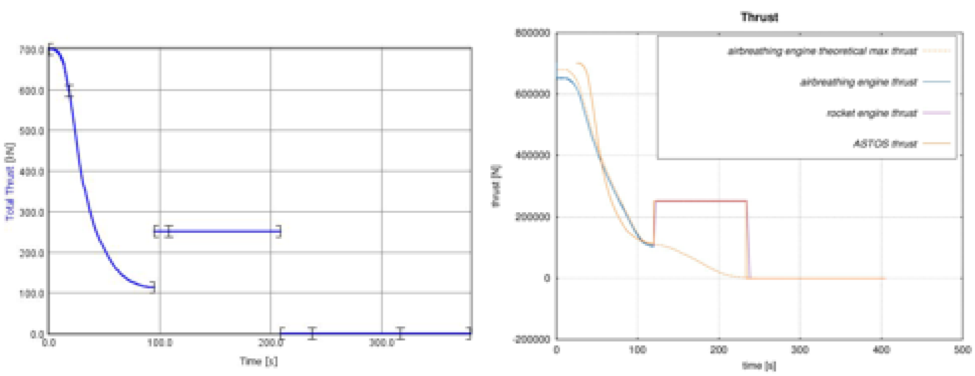


Figure 137: Thrust model with ASTOS (left) and comparison with the thrust modelling implemented within the ad-hoc built-in tool (right)

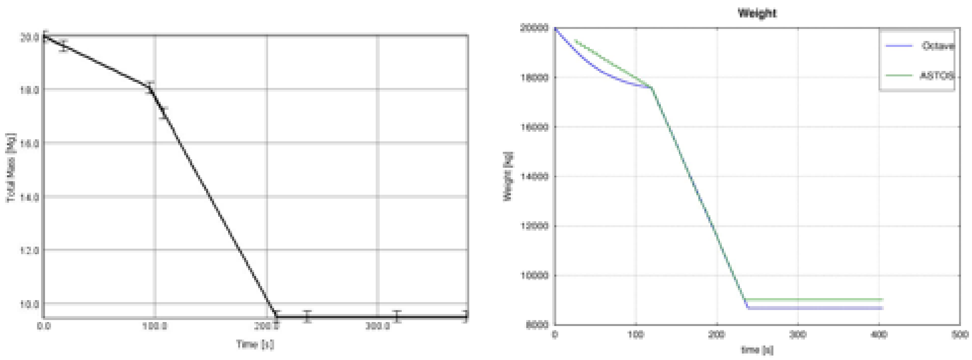


Figure 138: Wet mass reduction in ASTOS (left) and comparison with the model implemented within the ad-hoc built-in tool (right)

References

Allen H.J., (1958) Hypersonic Flight and the Re-entry Problem. T*he Twenty-First Wright Brothers Lecture. Journal of the Aerospace Sciences*, Vol. 25, No. 4, April 1958.

De Vita, Francesco, et al. (2015) Assessment of Hypersonic Flights Operation Scenarios: analysis of launch and reentry trajectories, and derived top level vehicle system and support infrastructure concepts and requirements. *20th AIAA International Space Planes and Hypersonic Systems and Technologies Conference*., Glasgow (UK).

Falempin, François. Ramjet and Dual Mode Operation. RTO/AVT/VKI (Lecture Ser. Adv. on Propulsion Technol. for High-Speed Aircraft), 2007.

Fusaro, R., et al. (2017) Database on hypersonic transportation systems: A versatile support for the technology roadmap generation and conceptual design activities. *21st AIAA International Space Planes and Hypersonics Technologies Conference, Hypersonics 2017*, 16 p.

Fusaro, R., et al (2017b). Conceptual design of a crewed reusable space transportation system aimed at parabolic flights: stakeholder analysis, mission concept selection, and spacecraft architecture definition. *CEAS Space Journal*, 9 (1), pp. 5-34. DOI: 10.1007/s12567-016-0131-7

Harloff, Gary J., and Brian M. Berkowitz. (1988) HASA: Hypersonic Aerospace Sizing Analysis for the Preliminary Design of Aerospace Vehicles.

Hirschel, E. H. (2005). Basics of aerothermodynamics (p. 201). *Berlin etc: Springer.*

Hirschel, E. H., & Weiland, C. (2009). Selected Aerothermodynamic Design Problems of Hypersonic Flight Vehicles, *vol. 229. Progress in Astronautics and Aeronautics*, AIAA, Reston, VA, 371-373.

Kuczera, H., & Sacher, P. W. (2011). Reusable Space Transportation Systems. Reusable Space Transportation Systems. *Springer, 2011*. ISBN: 978-3-540-89180-2, 1.

McClinton, Charles R. (2007) "High speed/hypersonic aircraft propulsion technology development." AVT-150 RTO AVT/VKI *Lecture Series, von Karman Institute, Rhode St. Genèse*, Belgium (2007): 12-15.

Raymer, D. P., et al. (1990) "Supersonic STOVL-the future is now." *Aerospace America* 28.8 (1990): 18.

Raymer, D. P., (1988). The impact of VTOL on the conceptual design process. *Aircraft Design, Systems and Operations Conference*. 1988.

Raymer, D. P., (2012). Aircraft design: A conceptual approach, *American Institute of Aeronautics and Astronautics*. Inc., Reston, VA (2012): 21.

Weiland, C. (2014). Classification and Project Data of Space Vehicles. In Aerodynamic Data of Space Vehicles (pp. 23-39). *Springer Berlin Heidelberg.*

Chapter 5

Wing Design

5.1 Introduction to Wing Design activities

The wing design is usually considered as one of the first activities to carry out just after the estimation of the high level design characteristics of the aerospace vehicle, such as the maximum take-off and landing weights, the fuel mass, the wing surface, etc. Indeed, the wing should be considered as one of the most important component of a fixed-wing aircraft. Besides the fact that in the world of very complex and innovative transportation systems, and in particular in the field of hypersonic vehicles, there could be different design architectures in which an actual wing is not identifiable (these peculiar cases will be further investigated in the Chapter devoted to the fuselage design), the wing "concept" remains the central elements as far as the aerodynamic forces generation is concerned. Moreover, the wing should be considered at the beginning of the design process because it has a relevant impact on different other architectural elements, but it imposes strict constraint to some on-board systems integration.

As it will be clearly described in this chapter, trade-offs will be required in order to match and satisfy the highest possible number of stakeholder requirements. In particular, there will always be a trade-off between aerodynamics, structures, on board systems integration, weight and balance, stability and manoeuvrability, accessibility, maintainability and safety.

In view of this context, this Chapter aims at providing the reader with an organized and structured methodology to carry out the design of a generic wing, following a Systems Engineering approach. For this reason, before starting with the wing geometry definition, the starting point is the list of functionalities that the wing is usually guaranteeing. From this list, a set of Areas of Impact will be derived, detailed and used to define the Figures of Merit to be exploited in the several trade-offs. One of most important trade-off is the wing vertical location. In this case, a proper structured approach is proposed, allowing the selection of the best option on the basis of the requirements generated at the beginning of the design process, through the analysis of the stakeholders and their relative expectations.

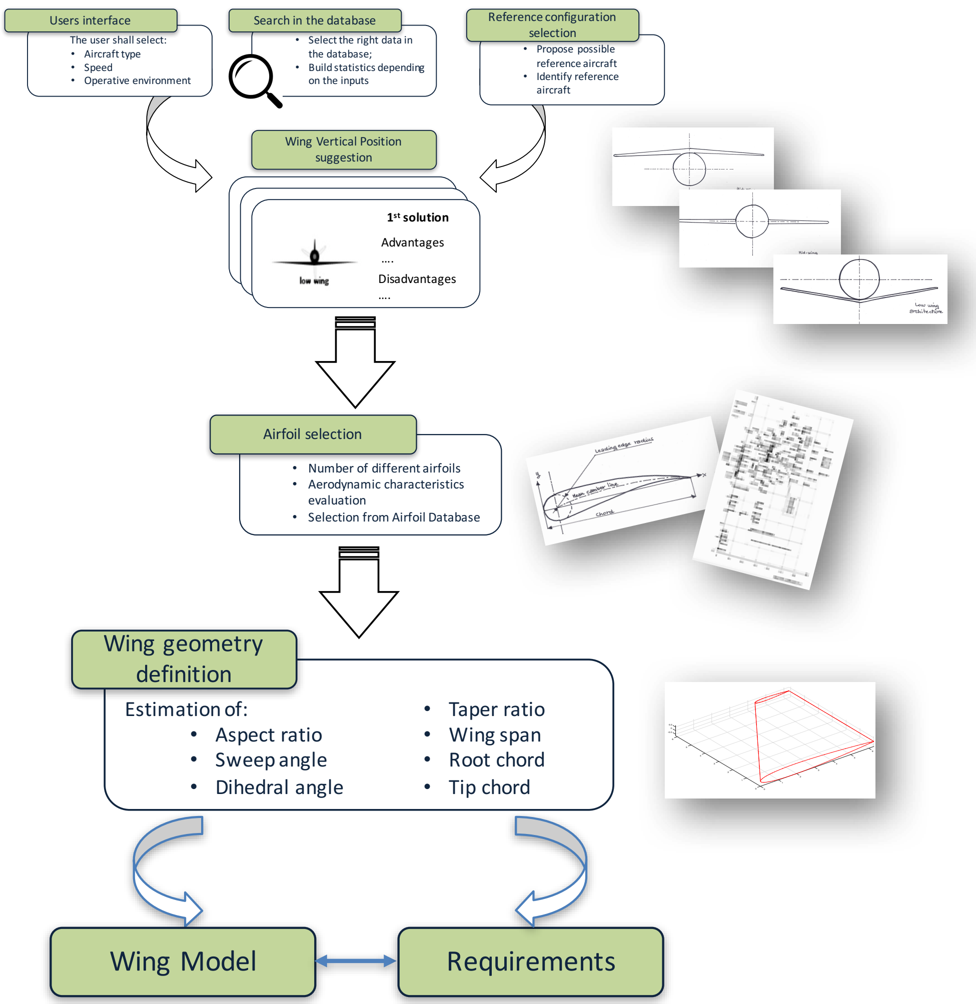


Figure 139: Overview of the Wing Design activity-flow

Moving to the specific aspects of the wing design, the process suggests to define the characteristics of the 2D airfoil at first and then moving to the 3D wing geometry. Figure 139 provides an overview of the major activities to be carried out in the wing design context. As it will be clearly described at the end of the chapter, a tool-chain has been properly built to support this design process, and guarantee a complete traceability of the requirements up to the ultimate geometrical characteristics.

5.2 Wing Functions

This chapter reports a list of functions that a general wing might perform.

- Primary Functions:
    - To generate sufficient lift force
    - To minimize the drag force
    - To minimize the pitching moment
    - To maximize the L/D ratio
- Secondary Functions:
    - To host fuel tanks
    - To host landing gear and relative actuation system
    - To host propulsive group
    - To host high lift devices
    - To host flight control surfaces and relative actuation system
    - To host Thermal Protection System
    - To guarantee a floating surface in case of splash down

Looking carefully at this list it is list of functions, it is clear that among the secondary functions, only a small subset should be guaranteed depending on the type of aircraft under design. In particular, it could be interesting to notice that in this list there are some typical aspects that should be taken into account when the designer is dealing with hypersonic transportation system, such as the capability of hosting Thermal Protection Systems or the need of maximizing the L/D ratio.

5.3 Number of wings

Besides the fact that in the last decades, only monoplane have been designed and developed, it would be important to understand advantages and disadvantages of three different aircraft categories:

- Monoplane
- Biplane
- Tri-wing plane

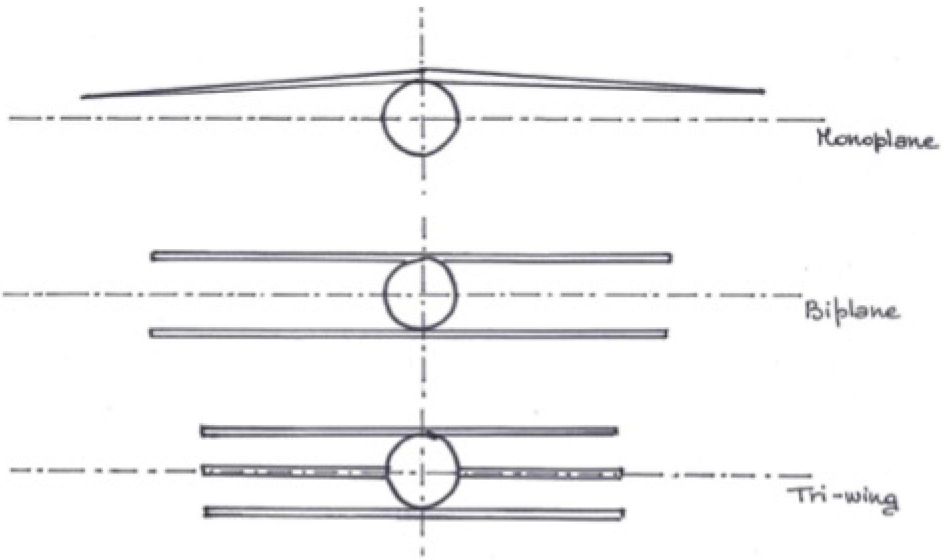


Figure 140: Monoplane, Biplane and Tri-wing aircraft configuration

First of all, it important to notice that a number of wings higher of three would not be practically feasible. The adoption of design architectures with more than a wing was crucially linked to manufacturing technology limitations, currently solved. Indeed, old manufacturing technologies were not so advanced to structurally support long wing, guaranteeing an adequate level of rigidity. The advancements in manufacturing technologies and materials in aerospace, and especially the advent of light aluminium alloys and innovative composite materials allowed filling the gap. Thus, once the technological obstacles have been overcome, the monoplane became almost the only practical option in conventional modern aircraft, despite of some benefits that would lead to the design of a vehicle with more than one wing.

The most significant benefit of having more than a wing is related to the aircraft controllability, considering that a fast rolling capabilities will be reached with a shorter wing span and this would match with lifting requirements only increasing the number of lifting surfaces.

Conversely, these options would have a higher weight and lower lifting capabilities with respect to a comparable single wing architecture. Moreover, the adoption of more wing would seriously limit the pilot visibility and reduce the maintainability of the overall aircraft.

It is also intuitive that for very high speed transportation, the only feasible architecture would be the monoplane architecture.

5.4 Wing Vertical Location

The wing vertical location is a major characteristic of the wing that is affected by the environment in which the aircraft will be operated, by its role and also by the speed regime. In view of this fact, we would like to introduce the following categorization criteria:

- Role:
  - Civil transportation
  - Military transportation
  - Fighter
  - Monitoring
- Speed regime
  - Subsonic
  - Supersonic
  - Hypersonic
- Environment
  - Lower atmosphere
  - Upper atmosphere
  - Inner space
  - Outer space

In-depth studying the possible impact of the different stakeholders on the selection of the most suitable wing vertical location, the following list of requirements has been elicited.

1. The volume available to host passengers shall be maximized.
2. The volume available to accommodate payload shall be maximized.
3. The landing gear weight shall be minimized.

4. The landing gear length shall be minimized.
5. The fuselage weight shall be minimized.
6. The number of cuts in the fuselage structure shall be reduced.
7. The aircraft configuration shall facilitate the loading and unloading operations.
8. The aircraft configuration shall guarantee the proper on-ground clearance to the ground.
9. The aircraft configuration shall facilitate refuelling operations.
10. The aircraft frontal section shall be minimized.
11. The ground effect shall minimized/maximized.
12. The aircraft configuration shall minimize the overall drag.
13. The aircraft configuration shall maximize the lifting capabilities.
14. The aircraft configuration shall enhance STOL capabilities.
15. The aircraft shall be operated from unprepared fields
16. The aircraft configuration shall guarantee a proper pilot visibility.
17. The aircraft configuration shall allow proper floating capabilities in case of emergency landing on waters.

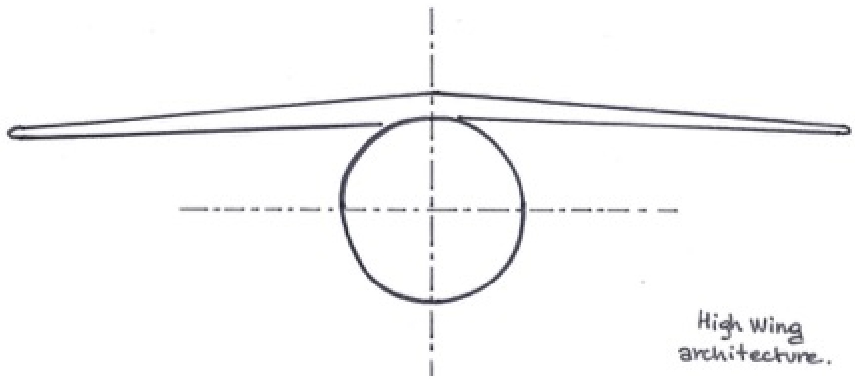


Figure 141: High Wing configuration

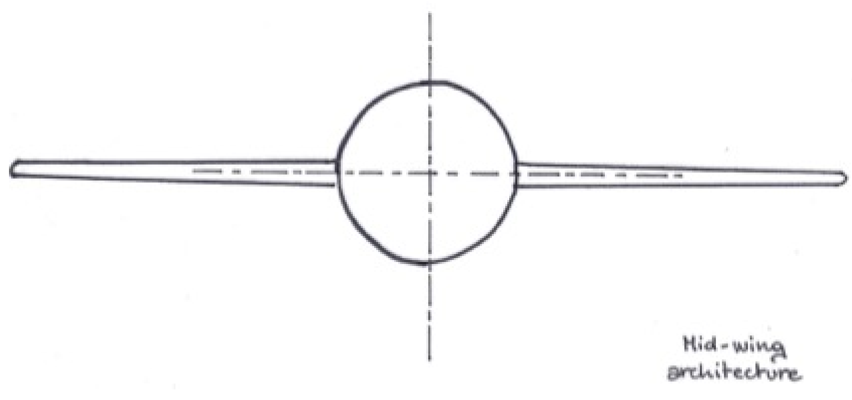


Figure 142: Mid Wing configuration

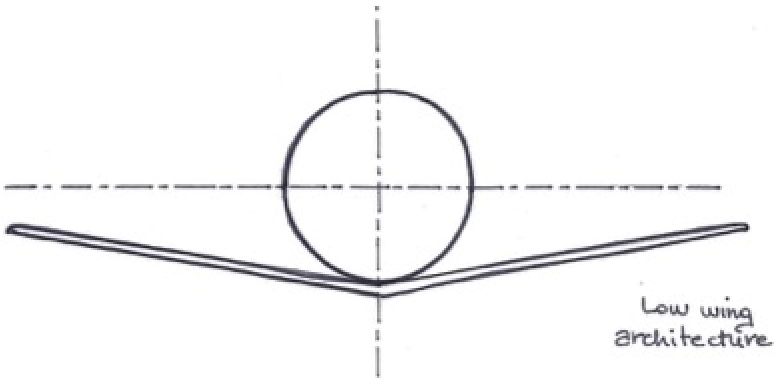


Figure 143: Low Wing configuration

Theoretically, three different options for the wing vertical positioning are feasible; high wing (Figure 141), mid wing (Figure 142) and low wing (Figure 143), where the adjectives refer to the relative placement with respect to the fuselage section. The following tables (Table 48, Table 49, Table 50) summarizes the major pros and cons of each of these solutions, with respect to different areas of interest.

From this analysis the following list of ***technical characteristics*** related to the several areas of impact previously analysed have been derived:

- Payload accommodation:
  - Volume available for payload
- Structure:
  - Wing weight and complexity
  - Fuselage weight and complexity
  - Landing Gear weight and complexity
- Logistic:
  - Passengers loading and unloading
  - Cargo loading and unloading
- Maintenance:
  - Systems accessibility
- Stability and Control:
  - Handling qualities in take-off
  - Handling qualities in climb
  - Handling qualities in cruise

- Handling qualities in re-entry
- Handling qualities in landing

Table 48: Pros and Cons of high-wing configuration

Wing Vertical Location *(with respect to the fuselage section)*	**Areas of Impact**	**Comments**
High	Payload Accommodation	• Enhanced volume for payload; both cargo and passengers would be easily accommodated.
	Structure	• Shorter and Lighter landing gear • Lighter wing in case of external struts. • Lighter fuselage due to the lower number of cuts and relative stiffened.
	Logistic and Maintenance	• Easy loading and unloading especially of cargo, because of the closest location of the fuselage to the ground. • Enhanced engine clearance (in case of wing-mounted engines) • Ground support infrastructure required to access to the engines (in case of wing-mounted engines) and to do the refueling.
	Aerodynamics	• Higher aerodynamic drag due to the enlarged frontal area and fairings. • Ground Effect reduced
	Stability and Control	• No special benefits
	Safety and Operations	• Larger wing flaps are required to guarantee STOL capability • Possibility of performing take-off from un-prepared fields

	• Limited pilot's visibility in case of small aircraft.

Table 49: Pros and Cons of mid-wing configuration

Wing Vertical Location *(with respect to the fuselage section)*	**Areas of Impact**	**Comments**
Medium	Payload Accommodation	• Limited room for the payload accommodation. • Enhanced possibility for carrying armaments under-wing
	Structure	• Increased weight of the fuselage due to the structural carrythrough.
	Logistic and Maintenance	• Advantages and disadvantages of mid wing configuration, as far as logistic and maintenance is concerned, will depend on the sizing of the aircraft. The relative distance of the wing from the ground will require special support tools.
	Aerodynamics	• Lowest drag configuration, because no external fairings are required.
	Stability and Control	• Increased maneuverability, also at high speed. • Enhanced aerobatic performances
	Safety and Operations	• Acceptable ground clearance • Best pilot visibility in small aircraft

Table 50: Pros and Cons of low-wing configuration

Wing Vertical Location *(with respect to the fuselage section)*	**Areas of Impact**	**Comments**
Low	Payload Accommodation	• Enhanced volume for payload; both cargo and passengers would be easily accommodated.
	Structure	• Reduced landing gear weight. • Landing gear can be easily stowed. • Dihedral angle required to enhance the ground clearance, increasing the wing complexity.
	Logistic and Maintenance	• Special ground equipment may be required to allow loading and unloading
	Aerodynamics	• Increase in fuel consumption , due to possible interference of propellers and wing
	Stability and Control	• No special benefits
	Safety and Operations	• The ground clearance can be enhanced, increasing the length of the landing gear legs. • Best option to guarantee possibility of escape in case of splash down, increasing the floating capabilities of the aircraft

Other considerations such as the aerodynamic characteristics or the impact on safety and operations are hardly quantifiable at this high level of design and for this reason, they are not part of the trade-off criteria but they are evaluated and linked to the selected configuration a-posteriori. It is clear that in order to carry out a rational trade-off to properly select the best wing positioning for a wing, it is necessary to evaluate (Table 51):

1. The impact of the technical and operational features on the wing vertical position (L_i, M_i, H_i). These evaluations are independent from the type of aircraft under investigation.
2. The relationship of the technical characteristics with the different aircraft categories. At this purpose, simple FoMs have been built, making use of statistical data easily derived by a Database.

Focusing on the impact of the technical and operational features on the wing vertical position (L_i, M_i, H_i), a voting process have been used on the basis of a 1 to 10 scale. The following tables (Table 52, Table 53 and Table 54) summarize the results of this process proposing possible values. The numerical values can be furtherly investigated through sensitivity analysis, tuning the models trying to reproduce existing case studies.

Table 51: Evaluation scheme suggested for the wing vertical location trade-off

	Importance of the characteristic for the product.	**Low wing**	**Mid wing**	**High wing**
Technical Feature 1	w_1	L_1	M_1	H_1
Technical Feature 2	w_2	L_2	M_2	H_2
Technical Feature 3	w_3	L_3	M_3	H_3
…..	…..	…...	…...	…...
		L	M	H

Table 52: Scoring rational for the wing vertical location trade-off (High-wing configuration)

Technical Feature	Importance for a High Wing configuration (h_i)	Comments
Volume available for payload	10 (8)	The highest value is related to the configuration with external structure. In case of internal carrythrough box, the volume available for payload can be moderately lower.
Wing weight and complexity	10 (7)	In case of external structure, the wing is light and with a low level of complexity. Complementary, in case of internal carrythrough box, the wing has a higher weight and complexity.
Fuselage weight and complexity	6 (5)	In case of external wing and the fuselage structure should not be interrupted, the fuselage increment in weight is only due to the presence of a heavier landing gear and of the aerodynamic fairings. Complementary, in case of internal carrythrough box, the fuselage weight is increased.
Landing gear weight and complexity	7	Medium weight and complexity landing gear can be envisaged in case of high wing configuration, mainly due to the possible wing

		installation.
Passengers Loading and Unloading	7	The high wing configuration allows passenger to access the aircraft without special problems. However, in case of internal carrythrough box, some comfort issues may arise.
Cargo Loading and Unloading	10	The high wing configuration is the best alternative from the logistic point of view. Indeed, the fuselage is closer to the terrain and the loading and unloading operations, for cargo is optimized.
System accessibility	8	This configuration diminishes the distance of the fuselage to the ground and this would facilitate the accessibility to many on-board systems. It is worth noticing that an intermediate weight as been assigned because there are also additional systems installed within the wing and thus, in this configuration, special on-ground equipment should be envisaged.
Handling qualities in take-off	7	The high wing configuration, both in case of internal and external mounting, can obstruct the pilot visibility during take-off phase.

Handling qualities in climb	4	The problem of pilot visibility is even more critical in climbing phase.
Handling qualities in cruise	5-8	The range of suggested values will strongly depend on the type of mountings. Indeed, this weighting factor is strictly related to the cross section area.
Handling qualities in re-entry	5	This configuration may suffer from serious injuries during due to the heating loads and the difficulties in providing an efficient Thermal Protection System. Of course, in case of vehicles that should be able to perform an orbit re-entry, external structures should not be considered.
Handling qualities in landing	10	The high wing configuration guarantees optimal controllability characteristics. Moreover, precision landing capabilities are increased thanks to the reduction of ground effect.

Table 53: Scoring rational for the wing vertical location trade-off (Mid-wing configuration)

Technical Feature	**Importance for a Medium Wing configuration (m_i)**	**Comments**
Volume available for payload	3	The mid wing configuration is the worst as far as the volume available to host payload is concerned.
Wing weight and complexity	7	The weight and complexity of this configuration is not so much affected.
Fuselage weight and complexity	6 (5)	The carrythrough box may imply an additional complexity and weight due to the required reinforcements.
Landing gear weight and complexity	7	A medium complexity landing gear can be envisaged for this application, with possible instalment both in fuselage and in wing.
Passengers Loading and Unloading	4	The medium wing can hampered the passengers loading and unloading operations.
Cargo Loading and Unloading	4	The medium wing can hampered the cargo loading

		and unloading operations.
System accessibility	5	The system accessibility in this case is strictly dependent on the system installation
Handling qualities in take-off	8	The mid-wing position is the optimal configuration from the aerodynamic point of view.
Handling qualities in climb	10	
Handling qualities in cruise	10	
Handling qualities in re-entry	7	The mid-wing configuration is not an optimal solution for re-entry phase.
Handling qualities in landing	8	No special problems are envisaged for the landing phase.

Table 54: Scoring rational for the wing vertical location trade-off (Low-wing configuration)

Technical Feature	Importance for a High Wing configuration (l_i)	Comments
Volume available for payload	10 (8)	The highest value is related to the configuration with external structure. In case of internal carrythrough box, the volume available for payload can be moderately lower.
Wing weight and complexity	8 (7)	In case of external structure, the wing is light and with a low level of complexity. Complementary, in case of internal carrythrough box, the wing has a higher weight and complexity
Fuselage weight and complexity	9 (8)	In case of external wing and the fuselage structure should not be interrupted.
Landing gear weight and complexity	9	A short landing gear can be envisaged. Thus, a reduction weight can be envisaged.
Passengers Loading and Unloading	8	Passengers loading and unloading does not face with special problems.

Cargo Loading and Unloading	8	Cargo loading and unloading does not face with special problems.
System accessibility	8	Systems accessibility does not face with special problems.
Handling qualities in take-off	6	Pilot Visibility, especially in case of the vertical take-off is obstruct.
Handling qualities in climb	8	No specific problems may be envisaged.
Handling qualities in cruise	8	No specific problems may be envisaged.
Handling qualities in re-entry	10	This configuration is the optimal one in terms of re-entry performances.
Handling qualities in landing	4	Low wing configuration hampered the visibility in landing phases.

Complementary, Table 55 suggests possible formulations to mathematically evaluate the impact of the identified technical characteristics on the different aircraft categories, i.e. the so-called weighting factors

Table 55: Weighting factor estimation for wing vertical location identification

Technical Feature	Mathematical formulation	Comments
Volume available for payload	$w_1 = \frac{V_{payload}}{L_{fus}\, A_{fus}}$	where $V_{payload}$ is the volume available for passengers and cargo [m^3]. L_{fus} is the length of the fuselage [m]. A_{fus} is the fuselage section area [m^2] This formula allows to estimate the available the volume efficiency for the different aircraft.
Wing weight and complexity	$w_2 = \frac{m_{wing}}{MTOM}$	m_{wing} is the wing mass estimation [kg]. $MTOM$ is the Maximum Take-Off Mass [kg] This formula allows estimating the relevance in terms of mass and complexity of the wing on the overall vehicle architecture.
Fuselage weight and complexity	$w_3 = \frac{m_{fus}}{MTOM}$	m_{fus} is the fuselage mass estimation [kg]. $MTOM$ is the Maximum Take-Off Mass [kg] This formula allows estimating the relevance in terms of mass and complexity of the fuselage on the overall vehicle architecture.
Landing gear weight and complexity	$w_4 = \frac{m_{lg}}{MTOM}$	m_{lg} is the landing gear mass estimation [kg]. $MTOM$ is the Maximum Take-Off Mass [kg] This formula allows estimating the relevance in terms of mass and complexity of the landing gear on the overall vehicle architecture.
Passengers Loading and Unloading	$w_5 = \frac{m_{pax}\ \cdot t_{load}}{MTOM \cdot\ TAT}$	$m_{pax}$is the passengers mass [kg]. t_{load} is the time estimated to perform the boarding/unboarding of passengers [s]. $MTOM$ is the Maximum Take-Off Mass [kg] TAT is the Turn Around Time [s] This formula allows estimating the impact of passengers loading and un-loading operations on the overall

		mission.
Cargo Loading and Unloading	$w_6 = \dfrac{m_{cargo} \cdot t_{load}}{MTOM \cdot TAT}$	m_{cargo} is the payload mass [kg]. t_{load} is the time estimated to perform the boarding/unboarding of cargo [s]. $MTOM$ is the Maximum Take-Off Mass [kg] TAT is the Turn Around Time [s] This formula allows estimating the impact of cargo loading and un-loading operations on the overall mission.
System accessibility	$w_7 = \dfrac{m_{sys} \cdot MTTR}{MTOM \cdot TAT}$	m_{sys} is the on-board systems mass [kg]. $MTTR$ is the time estimated to perform the maintenance actions after each single mission[s]. $MTOM$ is the Maximum Take-Off Mass [kg] TAT is the Turn Around Time [s] This formula allows estimating the impact of systems on the overall accessibility and maintenance characteristics of the aircraft.
Handling qualities in take-off	$w_8 = \dfrac{t_{TO} \cdot T_{TO}}{t_{mission} \cdot T_{max}}$	t_{TO} is the duration of the take-off maneuver [s] T_{TO} is the thrust required to perform the take-off [N].. $t_{mission}$ is the overall mission duration [s] T_{max} is the maximum available thrust [N]. This formula allows estimating the importance of take-off phase on the overall mission.
Handling qualities in climb	$w_9 = \dfrac{t_{climb} \cdot T_{climb}}{t_{mission} \cdot T_{max}}$	t_{climb} is the duration of the climb maneuver [s] T_{climb} is the thrust required to perform the climb phase [N].. $t_{mission}$ is the overall mission duration [s]

		T_{max} is the maximum available thrust [N]. It has to be noticed that in case of multi staged climb, performed with different propulsion systems, the overall FoM values should be evaluated as a $\sum_i \; w_{9_i}$. This formula allows estimating the importance of climb phase on the overall mission.
Handling qualities in cruise	$w_{10} = \frac{t_{climb} \; \cdot T_{climb}}{t_{mission} \cdot T_{max}}$	t_{climb} is the duration of the cruise maneuver [s] T_{climb} is the thrust required to perform the cruise phase [N].. $t_{mission}$ is the overall mission duration [s] T_{max} is the maximum available thrust [N]. This formula allows estimating the importance of cruise phase on the overall mission.
Handling qualities in re-entry	$w_{11} = \frac{t_{re} \; \cdot T_{re}}{t_{mission} \cdot T_{max}}$	t_{climb} is the duration of the re-entry maneuver [s] T_{climb} is the thrust required to perform the re-entry phase [N].. $t_{mission}$ is the overall mission duration [s] T_{max} is the maximum available thrust [N]. This formula allows estimating the importance of re-entry phase on the overall mission.
Handling qualities in landing	$w_{12} = \frac{t_{land} \; \cdot T_{land}}{t_{mission} \cdot T_{max}}$	t_{climb} is the duration of the land maneuver [s] T_{climb} is the thrust required to perform the land phase [N].. $t_{mission}$ is the overall mission duration [s] T_{max} is the maximum available thrust [N]. This formula allows estimating the importance of re-entry phase on the overall mission.

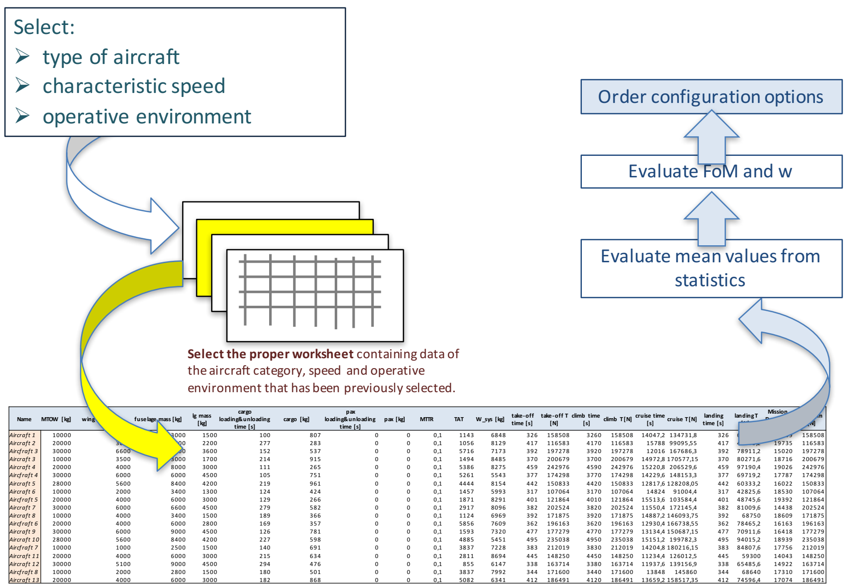


Figure 144: Pictorial view of the wing vertical location selection process

5.5 Airfoil Selection

Before going into the detail of the geometry definition, it is important to define the airfoil. Two different options would be followed:

1. Design a new airfoil (ad-hoc for a specific application) starting from aerodynamic characteristics directly required by the stakeholders or imposed by the mission profile.
2. Select existing airfoil on the basis of the envisaged aerodynamic characteristics.

The first case would be quite demanding, especially at the beginning of the design process, but of course it would be a mandatory step in case there is no existing airfoil able to satisfy the requirements. This would easily happen in case of innovative aircraft, like in hypersonic domain. However, also in these peculiar cases, in the framework of conceptual design, it would be better trying to identify possible reference airfoil to start the wing geometry definition and then, during the following design phases, improve the 2D profile, with the help of fluid-dynamics computations and aerothermodynamics evaluations.

In both cases, it is important to identify the most convenient design point within the flight envelope. In case of regular flight operations, consisting of take-off, climb, cruise, possible manoeuvres, cruise, descent and landing, it would be convenient to set the optimum function of the airfoil in cruise, considering that is the mission phase in which the aircraft spent the majority of its mission time. Considering special mission profiles envisaged for hypersonic transportation systems, different design points would be investigated, taking into account the fact that re-entry phase is really demanding and crucial, especially if reusable launchers would be considered.

5.5.1: Airfoil geometrical characteristics

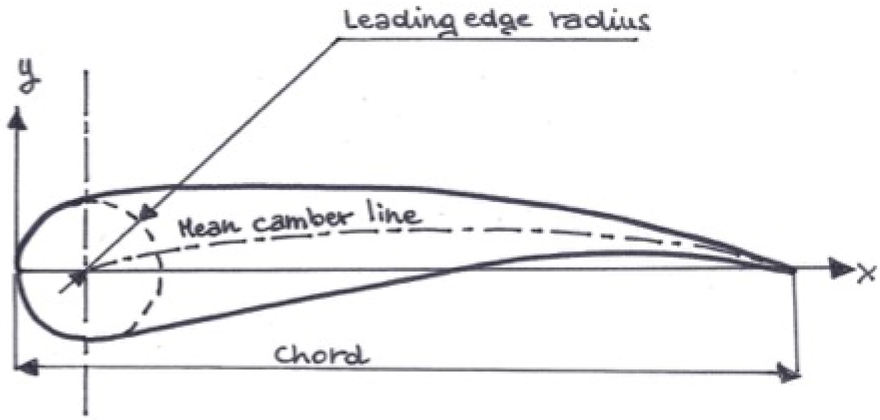


Figure 145: Airfoil main geometrical features

Either in the case of the design of a new airfoil or in the case of the selection of an existing one, it is very important to have a common understanding of the different definitions used. In particular, it is important to set a proper reference system. Usually, as it is highlighted in Figure 145, the origin of this system is simply the point on the airfoil that is farthest away from the trailing edge. On this basis is possible to define the chord of the 2D airfoil such as the straight line connecting the leading edge with the trailing edge, along the horizontal reference axis.

Taking a look to the different characteristics of a 2D wing section, the leading edge radius, the camber and the thickness distribution, are the three parameters that will allow categorizing the airfoil.

5.5.1.1 Leading Edge

Leading edge radius: it is the front part of the airfoil that is tangent to the upper and lower surfaces. Table # will allow the designer having an idea of the possible design consequences of the selection of a value for the leading edge radius. In particular, looking at the list of pros and cons, it could be immediately noticed that the trade-off is mainly influenced by the desired aerodynamic characteristics. Furthermore, it is clear that a sharp a trailing edge, with a radius tending towards zero would be the only feasible option for a hypersonic transportation system. In this specific case, the so-called diamond profiles will be employed.

This is a brief list of requirements with a possible impact on the selection of a proper leading edge radius.

1. The aircraft should be able to operate at high angle of attacks.
2. The wing contribution to the overall drag shall be minimized during the cruise speed.
3. The aircraft lifting capabilities during take-off and landing operations shall be maximized.
4. The wing configuration shall prevent from bow shock formation.

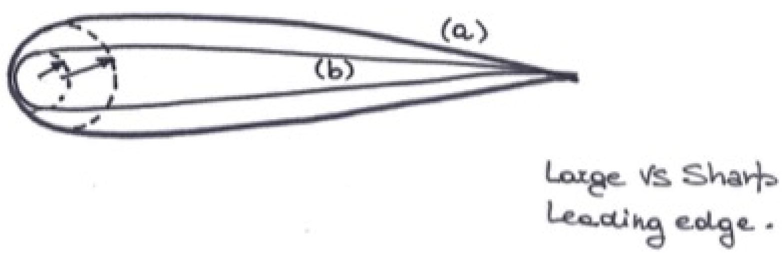


Figure 146: Large vs Sharp leading edge

Table 56: Pros and cons of large and sharp leading edge

Leading Edge radius	Pros	Cons
Large	• helps the air stay attached at the higher angles of attack • stall angle increased • enhanced lift in take-off and landing	• enhanced aerodynamic drag, especially at high speed
Sharp	• Prevent from bow shock formation. • Suitable for supersonic and hypersonic aircraft.	• Stall angle is decreased • Reduced lift in take-off and landing phases.

5.5.1.2 Camber

The camber characteristic refers to the curvature of the airfoil. From the geometrical point of view, it is possible to define the mean camber line that a line equidistant from the upper and lower surfaces and also the total airfoil camber, i.e. the maximum distance of the mean camber line from the chord line, as a percentage of the chord. Thinking of suggestions for hypersonic vehicles, a limited camber value will be adopted, aiming at obtaining the best aerodynamic performances keeping under control the aero-thermodynamic heating. A flat bottom wing will be preferred especially for those configurations with a demanding high speed cruise or re-entry phase.

This is a brief list of requirements with a possible impact on the selection of a proper airfoil camber.

1. The aircraft lower surface shall be as flat as possible, preventing from aero-thermodynamic issues.
2. The wing shall be able to generate a TBD amount of lift at zero angle of attack.
3. The wing contribution to the overall drag shall be minimized during the cruise phase
4. The wing lifting performances shall be maximized during the cruise phase.

5. The aircraft pitching moment shall me maximized.

Table 57: Pros and cons of different camber

Camber	Pros	Cons
Flat bottom airfoil	• Flat bottom wing surface, diminishing the aero-thermodynamic heating.	• Obsolete and used in at the very beginning of the fight era.
Double cambered	• Lift generation at zero angle of attack • Increased maximum lift of the airfoil	• Increase in drag • Increased pitching moment

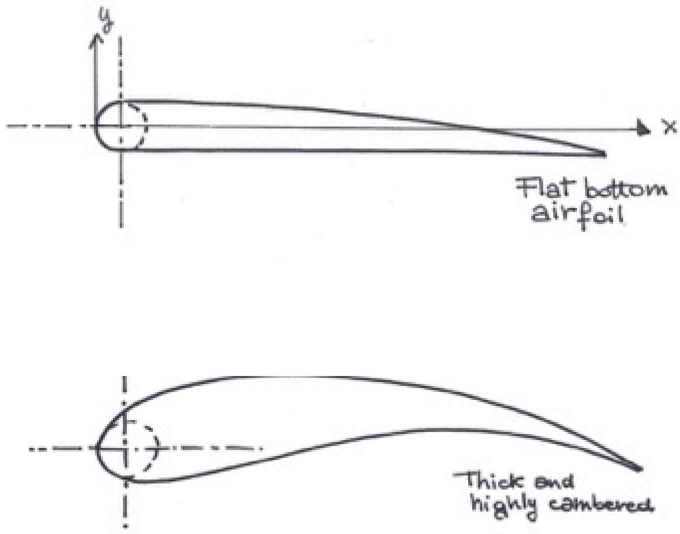


Figure 147: Different camber configurations

5.5.1.3 Airfoil Thickness

The thickness distribution of the airfoil can be defined as the distance from the upper surface, measured perpendicular to the mean camber line and it is defined as a function of the distance from the leading edge. Commonly, the ratio thickness-over-chord is used. The t/c parameter can affect both aerodynamic and operational performances. Indeed, the possibility of having a higher internal wing volume will facilitate the integration of important subsystems such as, fuel tanks, under carriage, avionics equipment and so on.

This is a brief list of requirements with a possible impact on the selection of a proper airfoil camber.

1. The wing shall maximize the lift generation.
2. The wing lifting performances shall be maximized during the cruise phase.
3. The wing contribution to the overall drag shall be minimized during the cruise phase.
4. The wing architecture shall maximize the available internal volume to maximize the room for systems integration.

Table 58: Pros and cons of thin and fat profiles

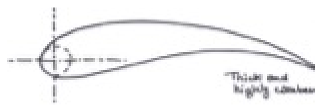
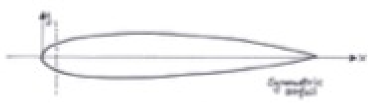
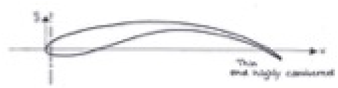
Thickness	Pros	Cons
Thin profile	• Good aerodynamic characteristics	• Limited available internal wing volume
Fat profile	• Not optimal from the aerodynamic point of view.	• Enhanced free internal wing volume.

The identification of a reasonable thickness distribution of the airfoil is only a first attempt that should be refined considering a 3D wing and some additional requirements such as the need of integrating some on-board subsystems or guaranteeing enough free volume for accessing the main equipment and doing maintenance. Moreover, some special cases would be observed in case of innovative configurations, such as in case a part of the propulsive system should be completely or partially integrated within the wing.

Table 59: Suggested values for (t/c) parameter

	Suggested value (Sadraey, 2012)	Comments
Low subsonic speed	$0.15 < (t/c)_{max} < 0.18$	Usually, high lift requirement is required for cargo transportation systems
High subsonic speed	$0.09 < (t/c)_{max} < 0.12$	Low lift requirements, as in case of passengers transportation
Supersonic speed	$0.03 < (t/c)_{max} < 0.09$	Aerodynamic requirements are stricter.
Hypersonic speed	$(t/c)_{max} < 0.03$	Aerodynamic requirements are stricter.

Table 60: Different airfoil configurations depending on the thickness distribution

Geometry	Sketch
Thick and highly cambered	
Symmetric	
Thin and highly cambered	
Supersonic double wedge	

Combining these features, several options may be theoretically derived. However, the most commonly used airfoil geometries are reported in the previous table.

5.5.2: Guide to the selection of the most suitable airfoil

This paragraph aims at providing the reader with a guide so select the most suitable airfoil following simple steps that can guide the designer from the stakeholders' requirements to the geometry definition. (Sadraey, 2012). To make this approach appealing and useful for the conceptual design activities, it is based on the following hypotheses:

1. the wing designer is planning to select the best airfoil from an existing airfoil section database. (NACA and Eppler are some possible examples).
2. The major characteristics of the aircraft have been already estimated in previous design steps; In particular, it is important to have idea of the average aircraft weight, W_{mean} (under the assumption of selecting the best profile optimized for the cruise phase), the cruise speed, V_C, a and the wing surface, S.

The selection process consists of less than ten steps that are summarized and commented in Table 61. As it is possible to notice from the sequence of proposed estimations, it is a typical top-down approach. Indeed, starting from high level lift estimation at aircraft-level, the problem is decomposed until identifying the airfoil contribution.

Table 61: Step-by-step approach for the selection of a suitable airfoil.

Step	Formulas	Comments
Calculate the aircraft ideal cruise lift coefficient	$$C_{L_C} = \frac{2W_{mean}}{\rho V_C^2 S}$$ *Where:* C_{L_C} is the aircraft ideal cruise lift coefficient; ρ is the air density (at cruise altitude) [kg/m^3]; $S$is wing surface [m^2].	This first step allows estimating a first value for the requirements of the overall aircraft in an intermediate point of the cruise.
Calculate the wing lift coefficient	$$C_{L_{C_W}} = \frac{C_{L_C}}{k_w}$$ Where: $C_{L_{C_W}}$ is the wing cruise lift coefficient; k_w is the wing contribution percentage to the overall aircraft lifting characteristics.	This step allows the designer to move from aircraft-level to the wing-level. Considering that the wing is usually the solely responsible for the generation of the lift, k_w can be set at 0.95 for traditional configuration [REF SE]. It is clear that in case of configuration on which tail/canard surfaces or the fuselage are more strongly contributing to the overall aircraft lifting capacity, this value should be properly reduced.
Calculate the wing airfoil ideal lift coefficient	$$C_{L_i} = \frac{C_{L_{C_W}}}{k_a}$$ C_{L_i} is the wing cruise lift coefficient; k_a is the wing airfoil lifting contribution to the wing lifting coefficient.	This step allows moving from a 3D problem at wing level, to a 2D investigation, focusing on the airfoil. The parametric coefficient k_a present in this equation can be set at 0.9 in conceptual and preliminary design evaluation. This allows considering the fact that the wing span is limited, and the possible presence of sweep angle and non-constant chord.
Calculate the aircraft	$$C_{L_{max}} = \frac{2W_{TO}}{\rho_0 V_S^2 S}$$	This step is absolutely similar to the very first one, but allows deriving the maximum aircraft lift coefficient.

maximum lift coefficient	*Where:* $C_{L_{max}}$ is the aircraft maximum lift coefficient; ρ_0 is the air density (at sea level) [kg/m^3]; $S$is wing surface [m^2]. $W_{TO}$is the maximum take-off weight; V_S is the stall speed [m/s]	Following the same top-down approach it will be possible to estimate the wing airfoil maximum lift coefficient.
Calculate the wing maximum lift coefficient	$C_{L_{maxW}} = \dfrac{C_{L_{max}}}{k_w}$ Where: $C_{L_{maxW}}$ is the maximum wing lift coefficient; k_w is the wing contribution percentage to the overall aircraft lifting characteristics.	
Calculate the wing airfoil gross maximum lift coefficient	$C_{l_{max-gross}} = \dfrac{C_{L_{maxW}}}{k_a}$ $C_{L_{max-gross}}$ is the wing airfoil gross maximum lift coefficient ; k_a is the wing airfoil lifting contribution to the wing lifting coefficient.	The effect of High Lift Devices (HLD) is included
Calculate the wing airfoil net maximum lift coefficient	$C_{l_{max}} = C_{l_{max-gross}} - \Delta C_{l_{HLD}}$	Where the contribution to the to the wing maximum lift coefficient depends on the geometry, type and maximum deflection of the selected HLD.

Identify the airfoil selection alternatives that deliver the desired C_{l_i} and C_{lmax}

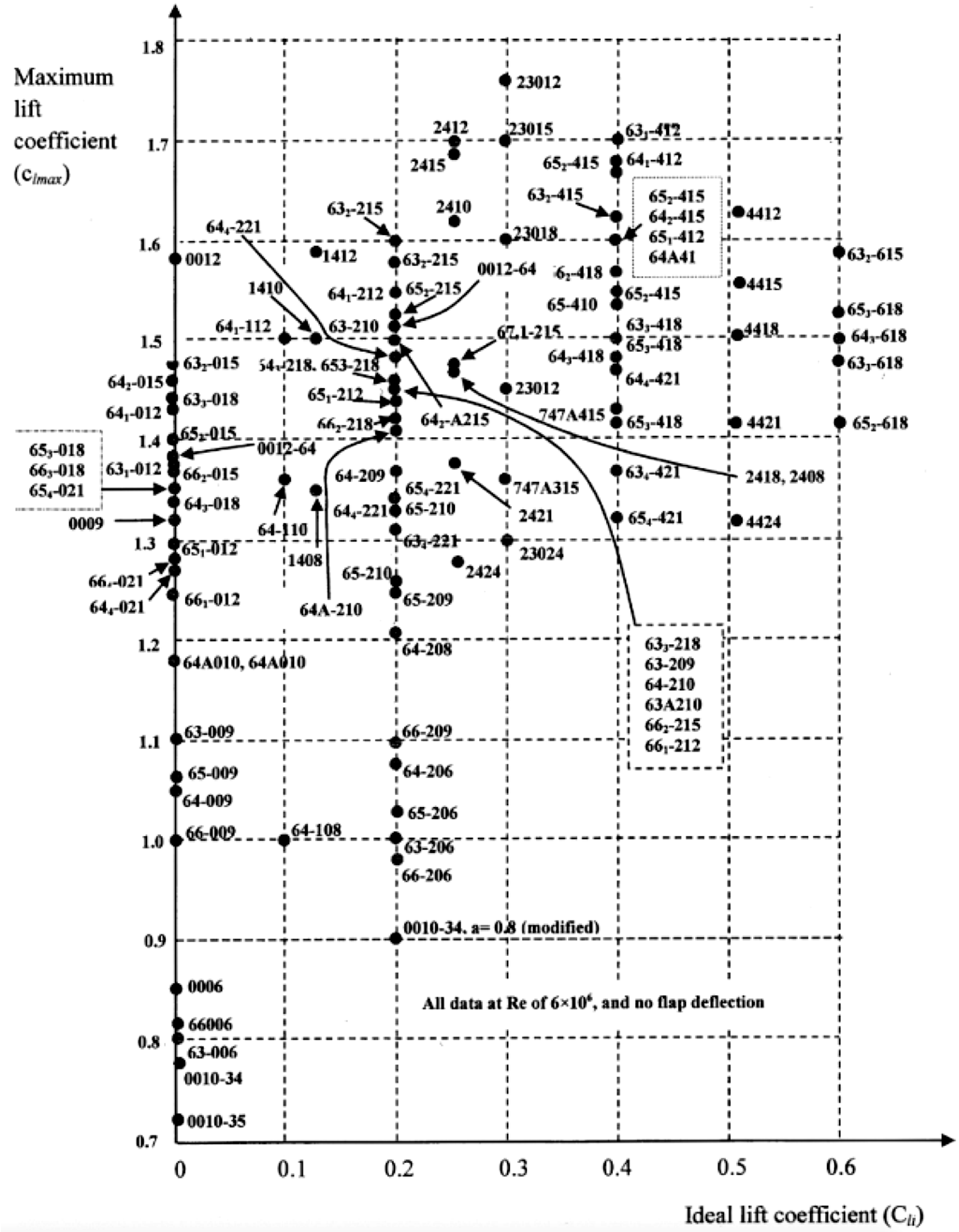


Figure 148: Maximum Lift coefficient vs Ideal Lift coefficient (Sadraey, 2012)

Figure 148 reports an example of airfoil database that could be used to select the most suitable airfoil.

In particular, following the Table 61, it is possible to select a proper subset of suitable airfoil. However, the selection of the best one will be carried out considering one or more optimization criteria. In particular, following the suggestions given by literature, it will be possible to select the best airfoil among the identified family, looking at that one with the lowest $(t/c)_{max}$.

Once the airfoil has been selected, it is possible to proceed with the definition of the geometry of the airfoil.

5.6 Wing Geometry Definition

5.6.1 Wing Incidence

One of the first parameters that should be selected at the beginning of the wing geometry definition procedure is the wing incidence. Referring to the literature, this parameter could be defined as the angle between the fuselage centre line and the wing root chord. In literature, this angle is also referred to wing settling angle (α_{set}).

Two different architecture alternatives can be envisaged: a variable wing incidence and a fixed one. Possible pros and cons for both these options from different perspectives, have been evaluated and reported in Table #. In the end it is possible to convene that a fixed wing incidence is the best option to reduce weight and to avoid possible huge safety and operational constraint. This is even more true if hypersonic transportation systems are concerned. Indeed, the possibility of changing the incidence of very big surfaces at very high speed would require a too huge amount of power in front of a very limited aerodynamic and stability advantage.

Table 62: Comparison between fixed and variable wing incidence configurations

Wing Incidence type	Pros	Cons
Fixed	Easy maintenance Easy operations Less structural weight Easy integration of the wing with the fuselage Safer	Cannot be optimized all along the mission
Variable	Aerodynamically optimized Enhanced stability	Increased structural weight Reduced available internal volume Increased power consumption Complex maintenance Complex operations requiring ad hoc controls

Hypothesizing that a fixed wing incidence has strategy has been selected, in order to understand how to select the best value of α_{set}, it is important to start listing which high level requirements can have a deeper impact on this parameter:

Aerodynamic

1. The wing shall maximize the lift generation.
2. The wing lifting performances shall be maximized during the cruise phase.
3. The wing contribution to the overall drag shall be minimized during the cruise phase
4. During cruise, the drag generated by the fuselage shall be minimized, i.e, the fuselage angle of attack in cruise shall be null.

Operations

5. The available excursion of angle of attack during take-off operation shall be maximized.
6. During cruise phase, the fuselage angle of attack in cruise shall be null in order to guarantee the maximum comfort level.
7. The aircraft landing distance shall be minimized.

Taking into account all these requirements, a first way to select the best value of α_{set} is to exploit, if available, the airfoil lifting curve coefficient value. In this case the wing settling angle shall correspond to that angle for which the selected airfoil is able to generate the ideal lift coefficient.

In case the confidence level in the airfoil aerodynamic data would be limited, a statistical approach may be implied. In particular, it would be possible to estimate α_{set} using the following equation:

$$\alpha_{set} = (\alpha_{set})_0 - \Delta i_w$$

where:

$(\alpha_{set})_0$ can be identified following a statistical approach and it is strictly related to the type of aircraft. Please notice that some useful first value attempts are reported in Table 63.

Table 63: Suggestions for wing incidence estimation.

Aircraft Type	Wing Incidence
Supersonic fighters	0 – 1 deg
Hypersonic Transportation Systems	0 – 1 deg
General Aviation	2 – 4 deg
Jet transportation	3 – 5 deg

Δi_w is a correction that takes into account corrections due to operational constraints such as the presence of cargo transportation with aft cargo doors or the possible need of improving the stopping performances during landing operations, maximising the weight on the braked wheels.

5.6.2 Aspect Ratio

The Aspect Ratio is an important characteristic of the wing and has a relevant impact at aircraft level. The most generic definition of AR is the ratio between wing span and wing MAC.

$$AR = \frac{b}{c_{MAC}}$$

In case of rectangular wing, AR can also be evaluated as follows:

$$AR = \frac{b^2}{S}$$

In order to define the best AR, this paragraph summarizes the impact of AR on the different design areas. In particular, for each area, a list of requirements that will impact on the selection of the best value of AR is reported. Then, a table will summarize the impact of the AR on several performances related to the different areas of interest.

List of requirements that can have an impact on the selection of the AR.

1. The wing shall maximize the lift generation
2. The wing geometry shall minimize the 3D effect due to wing tip vortex.
3. The wing stall shall be avoided.
4. The wing shall be able to maximize the lift-over-drag ratio.
5. The wing stall shall be postponed.
6. The tail stall shall be postponed after wing stall
7. The wing weight shall be reduced.
8. The wing production cost shall be reduced
9. The wing geometry shall maximize the effectiveness of wing control surfaces.
10. Gliding performances shall be maximized.

Table 64: Impact of AR on aerodynamic performances

Aerodynamic performance	**Effect of the increment of AR**	
	Pros	**Cons**
$\mathbf{C_L^{3D}}$	C_L^{3D} keeps closer to C_L^{2D} because wing tip effects are reduced.	
$\mathbf{C_{L\alpha}}$	$C_{L_\alpha} = \frac{C_{l_\alpha}}{1+\frac{C_{L_\alpha}}{\pi AR}}$ is increased	
$\mathbf{C_{Lmax}}$	The maximum wing lift coefficient is increased.	
$\left(\frac{L}{D}\right)_{max}$	The maximum lift-over-drag ratio is increased because	

	$\left(\frac{L}{D}\right)_{max} = f(\sqrt{AR})$	
C_{Di}	The induced drag is decreses, due to the fact that: $C_{D_i} = \frac{C_L^2}{\pi\, e\, AR}$	

Table 65: Effect of AR on structural and manufacturing wing characteristics.

Structural and manufacturing characteristics	**Effect of the increment of AR**	
	Pros	**Cons**
W_{empty}		The increase in AR implies an increase in bending moment and this will imply the need of additional structural reinforcements. Moreover, additional reinforcements are required in order to counteract the y-axis wing stiffness reduction.

Table 66: Effect of AR on stability and manoeuvrability performances

Stability and maneuverability performance	Effect of the increment of AR	
	Pros	**Cons**
downwash	The increment of AR implies a reduction of the downwash effect with benefic impact on maneuverability, longitudinal stability and longitudinal control.	
Aileron effectiveness	The increment in AR implies an increment of the aileron arm, with a positive impact in lateral control.	
I_{xx}		The increment in AR implies an increment of the moment of inertia about x-axis. This can have a negative effect, reducing the maneuverability in roll.
Aileron reversal		Increasing the AR, there is an higher risk of facing with aileron reversal.

Table 67: Effect of AR on aircraft Weight&Balance

Weight and balance performance	Effect of the increment of AR	
	Pros	**Cons**
Fuel distribution and CG shift		With a higher AR, the wing fuel tanks integration and the fuel results to be more widely distributed. Looking at the overall aircraft balance, a higher AR implies a wider CG shift range.

Table 68: Effect of AR on wing costs

Cost performance	Effect of the increment of AR	
	Pros	**Cons**
Overall wing cost		The wing cost is increased in view of the fact that in order to build a wing with a higher AR additional structural reinforcements are required.

Table 69: Effect of AR on aircraft logistics

Logistics performance	Effect of the increment of AR	
	Pros	Cons
Difficulties in storing		The higher AR will implies additional logistical difficulties in parking and storing the aircraft.

Table 70: Effect of AR on Safety

Safety performance	Effect of the increment of AR	
	Pros	Cons
Gliding range	The gliding performances are improved with the adoption of a higher AR wing. This allows increasing safety in case of engine failures.	
α_S		Stall angle decreases in view of the wing effective angle of attack reduction. In particular, for safety recovery requirements it is convenient to set: $(AR)_{canard} > (AR)_{wing} > (AR)_{tail}$

On the basis of statistical analysis, (Raymer, 2012) tried to express the Aspect Ratio as function of the aircraft type and of the maximum Mach number. For hypersonic vehicles, considering the very limited number of projects and programs, estimation based on (Sadraey, 2012) is here proposed.

Table 71: Suggestions for AR estimations

Type of aircraft	Aspect Ratio estimation [REF Raymer]	Suggestion [REF SE]
Sailplane	$0.19\left(best\,\frac{L}{D}\right)^{\frac{1}{3}}$	20 – 40
Jet trainer	$4.737\ (M_{max})^{-0.979}$	4 - 8
Jet fighter	$4.110\ (M_{max})^{-0.0622}$	2 - 4
Military Cargo	$5.570\ (M_{max})^{-1.075}$	6 – 12
Low subsonic Transport		6 - 9
High subsonic Transport		8 – 12
Supersonic transport		2 - 4
Hypersonic transport		1 - 3

5.6.3 Wing Sweep angle

The wing sweep angle is defined as the angle between a constant percentage chord line along the semi-span of the wing and the lateral axis perpendicular to the aircraft centre line (y-axis). In particular, to be more precise, this is the definition of the Leading Edge sweep angle. In the same way, it is possible to define the Trailing Edge sweep angle as the angle between the wing trailing edge and the longitudinal axis of the aircraft, the quarter chord sweep as the angle between the wing quarter chord line and the longitudinal axis and finally the 50% chord sweep as the angle between the wing 50% chord line and the aircraft longitudinal axis.

Conventionally, in literature, a sweep angle is considered positive (aft sweep) whether the wing is inclined towards the tail; otherwise, it is referred to as forward sweep (negative).

Two different architectural alternatives should be evaluated:

- fixed wing sweep angle
- variable wing sweep angle.

Pro and cons of the two options have been in depth analysed. In particular, it has to be noticed that the variable geometry has been deeply investigated in the late 1980s especially because it offers the best compromise among very different mission phases. However, the high level of complexity, risk and costs related to this innovative and technologically advanced solution, forced the engineers to focus on different design architectures.

Moreover, as far as the wing configuration is concerned, it is possible to classify the alternatives in

- Single wing sweep angle
- Double sweep angle.

Considering these alternatives, a double wing sweep can be used to compensate variations for aerodynamic in low and high speed regimes and it would be very useful for single stage hypersonic vehicles that should face with flight phases with a wide range of speed and altitudes.

This is the list of requirements having the major impact on the selection of the proper wing sweep angle.

1. The wing area shall be included within the Mach cone to withstand the structural and heating loads.
2. The wing shall maximize the lift generation
3. The stall speed shall be increased.
4. Lateral stability shall be enhanced.
5. Lateral manoeuvrability shall be enhanced.
6. The aircraft controllability in turbulence shall be enhanced.

Considering the following Table, it is clear that aerodynamic and stability are the two important areas of interest with the major impact on aerodynamic and stability.

Table 72: Effect of wing sweep on aircraft aerodynamics

Aerodynamic performance	**Effect of the increment of Sweep angle**	
	Pros	**Cons**
Critical Mach number	The increment in sweep angle will allow to reduce the effective aerodynamic chord of a factor $1/\cos\Lambda$. This allows a reduction of the maximum airfoil thickness implying an enhancement in the critical Mach number.	.
Oswald factor		The presence of a sweep angle implies a reduction of the Oswald factor, showing the fact that the wing lift distribution is no more elliptic. Referring to (Sadraey, 2012),

		$e = 4.61(1 - 0.045\, AR^{0.68})[\cos(\Lambda_{LE})]^{0.15} - 3.1$
$(C_{Lmax})_{wing}$	The sweep angle has a beneficial effect on the maximum wing lifting coefficient.	
$(C_{Lmax})_{aircraft}$		The aircraft lifting coefficient is reduced due to a reduced controllability in pitch-up situations.
V_{stall}		The aircraft stall speed is increased due to a reduced controllability in pitch-up situations. The stall phenomena are more critical for the wing tip parts.
AR_{eff}	The presence of a certain sweep angle causes a reduction in the effective aspect ratio. This can have several pros and cons as reported in Table #.	

Table 73: Effect of wing sweep on aircraft stability performance

Stability performance	**Effect of the increment of Sweep angle**	
	Pros	**Cons**
Aircraft pitching moment	The wing aerodynamic center is moving aft and the aircraft pitching moment is increased due to the CG position forward of the aircraft aerodynamic center	

Spiral stability	The increment in sweep angle provides the aircraft with a negative rolling moment that increases the so called natural dihedral effect, providing the aircraft with a more spirally stable configuration.	
Dutch Roll		The increment in wing sweep angle implies a reduction in Dutch-roll damping ratio.
Turbulence resistance	High values of wing sweep angle, together with high wing loading provides good riding performances in turbulence.	
Lateral maneuverability	Lateral maneuverability is enhanced due reduction in the moment of inertia about the x-axis, due to the reduction of the effective wing aperture.	

In order to properly carry out a first estimation of a suitable wing sweep angle, the major constraint is the need of being compliant with the Mach cone aperture.

From the theoretical point of view, the semi-aperture of the Mach cone (μ) can be defined as

$$\mu = \sin^{-1}\left(\frac{1}{M}\right)$$

and the relative sweep angle can be usually defined as

$$\Lambda = k_{\Lambda}(90 - \mu)$$

where k_Λ is a factor that will be used to diminish the wave drag in supersonic and hypersonic speed. Considering some results provided by literature, a factor of 1.2 will guarantee the lowest wave drag, avoiding the shock wave to be very closed to the wing leading edge, generating high temperature due to a serious increment of the aerodynamic heating (Figure 150). Please notice that leading edge sweep angles greater than 90 deg do not have sense.

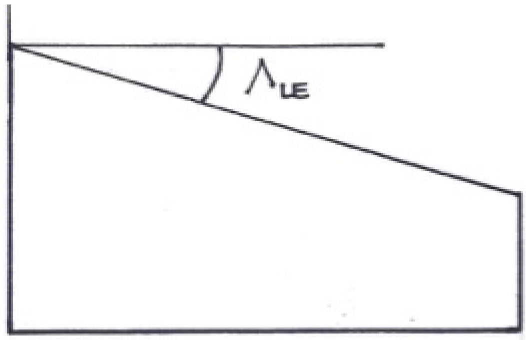


Figure 149: Wing Sweep angle definition

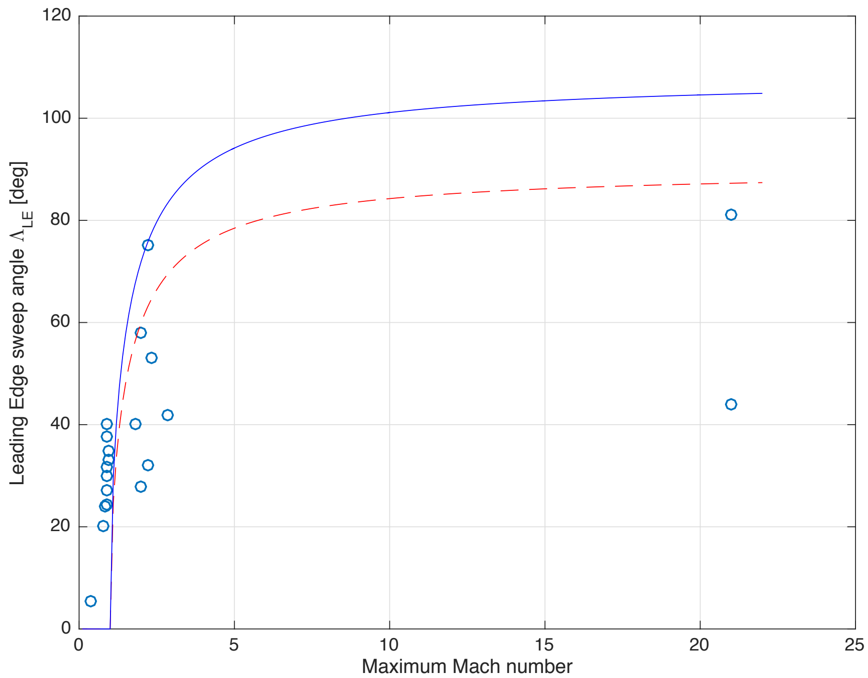


Figure 150: Leading edge sweep angle vs maximum Mach number.

5.6.4 Dihedral Angle

The dihedral angle is usually defined as the positive angle between the chord line plane of a wing with the xy plane. In case the wing tip is lower than the xy plane, this angle is called negative dihedral or anhedral angle.

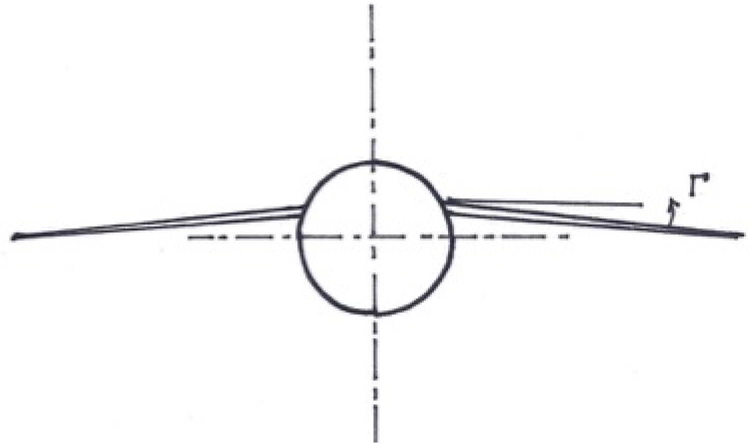


Figure 151: Dihedral angle definition

Different requirements may guide in selecting a proper value for the dihedral angle.

1. The wing shall maximize the lift generation
2. Lateral stability shall be enhanced.
3. Lateral manoeuvrability shall be enhanced.
4. The wing architecture shall guarantee proper clearance to ground and water.
5. The aircraft shall be easy to be maintained.
6. Logistic on-ground infrastructures required to perform maintenance shall be reduced.

As it evident from the results of the investigations summarized in the following table, the major difference among the two different wing architectures in terms of dihedral angle is strictly related to stability and control characteristics. A positive dihedral angle is beneficial for the lateral stability but it decreases the Duch-roll damping ratio. Thus, one of the most exploited way of approaching the wing design is to define the above mentioned parameters with a greater impact on the overall design and performances, such as the wing sweep angle(s) and the wing vertical location and then, in a following iteration, try to find the best dihedral angle to cope with stability, controllability and operations.

Table 74: Effect of Dihedral angle on aircraft stability performance

Stability and control performance	**Effect of Dihedral angle**	
	Pros	**Cons**
Lateral stability	A positive dihedral angle is beneficial for lateral stability	
Lateral Control		A positive dihedral angle decreases duch-roll damping coefficient.

Table 75: Effect of Dihedral angle on aircraft aerodynamic performance

Aerodynamic performance	**Effect of Dihedral angle**	
	Pros	**Cons**
Lift		The presence of a dihedral angle (both a positive or a negative one) will have a detrimental effect on the wing lifting capability. Indeed, the presence of a dihedral angle implies a reduction in the effective wing surface as a $f(\cos\Gamma)$, causing a lift reduction.

Table 76: Effect of Dihedral angle on logistics and operations

Operational and logistical performance	Effect of Dihedral angle	
	Pros	Cons
Clearance	The presence of a positive dihedral angle would increase the ground and water clearance.	
Logistics		The presence of a positive dihedral angle can enhance the operational efforts in carrying out maintenance simply considering the fact that on-ground support equipment will be used to reach higher distances from the ground floor
Fuel consumption strategy	In case of wing tanks, the presence of a positive dihedral angle naturally pushes the fuel towards the wing root, where usually collectors and pumps are located.	

Looking at existing aircraft configurations it is also important to notice that there is a close relationship between wing vertical location and the dihedral effect and this is mainly due to lateral stability and control requirements. As it is possible to be noticed in the following table, where some examples are reported, the presence of a high wing it is usually associated with a negative dihedral angle. Conversely, the presence of positive dihedral angles is mainly associated with a low wing configuration. In addition, for the same reasons, there is a strict relation with the sweep angle too. Referring to the practical suggestions provided in (Raymer, 2012), 10 deg of sweep provides about 1 deg of effective dihedral. In particular, in case of a forward swept wing, a negative dihedral angle will be required.

Table 77 (Sadraey, 2012) reports some useful ranges for dihedral angle values, allowing to carry out a first high level estimation of this peculiar wing characteristics on the bases of wing vertical location and sweep angle.

Table 77: Dihedral angle suitable range taking into account the mutual interactions between sweep angle and wing vertical location

	Low Wing	**Mid Wing**	**High Wing**
Un-swept	5 to 10 deg	3 to 6 deg	-4 to -10 deg
Low-subsonic swept	2 to 5 deg	-3 to 3 deg	-3 to -6 deg
High subsonic swept	3 to 8 deg	-4 to 2 deg	-5 to -10 deg
Supersonic swept	0 to -3 deg	1 to -4 deg	0 to -5 deg
Hypersonic swept	1 to 0 deg	0 to -1 deg	-1 to -2 deg

5.6.5 Taper Ratio

Wing taper ratio is defined as the ratio between the wing tip chord and the wing root chord.

$$\lambda = \frac{C_t}{C_r}$$

where, λ is the taper ratio, C_t is the chord measured at the wing tip and C_r is the chord measured at the wing root. Due to this definition, this parameter ranges from 0 in case of a pure delta wing to 1, in case of a more traditional rectangular wing.

This is the list of requirements that may affect the selection of a proper taper ratio.

1. The wing planform shall maximize the lift distribution.
2. The wing shall maximize the lift generation
3. The wing manufacturing costs shall be minimized.
4. The wing weight shall be minimized
5. Lateral manoeuvrability shall be enhanced
6. Lateral stability shall be enhanced.

The following table summarizes the major pros and cons of different taper ratios.

Table 78: Effect of taper ratio on aerodynamics

Aerodynamic performance	**Effect of taper ratio**	
	Pros	**Cons**
Lift distribution		The presence of taper ratio will deeply affect the wing lift distribution. In particular, the best lofting distribution (the elliptical one), can be obtained only with a unitary taper ratio.

Table 79: Effect of taper ratio on costs

Cost performances	**Effect of taper ratio**	
	Pros	**Cons**
Wing production		The presence of taper ratio can increase the production costs of a wing, requiring special tools or manufacturing processes.

Table 80: Effect of taper ratio on Weight&Balance

Weight and balance performances	Effect of taper ratio	
	Pros	**Cons**
Wing weight	The presence of taper ratio can reduce the wing weight because it can guarantee a lower bending moment at the wing root, thus, requiring less efforts in structural reinforcements.	

Table 81: Effect of taper ratio on Stability and Control performances

Stability and control performances	Effect of taper ratio	
	Pros	**Cons**
Lateral controllability	A tapered wing is characterized by a lower inertia about the x-axis and for this reason it results in having a higher lateral controllability.	
Spiral Stability	The presence of a tapered wing will augment the effect of an already present sweep angle, implying a higher spiral stability.	

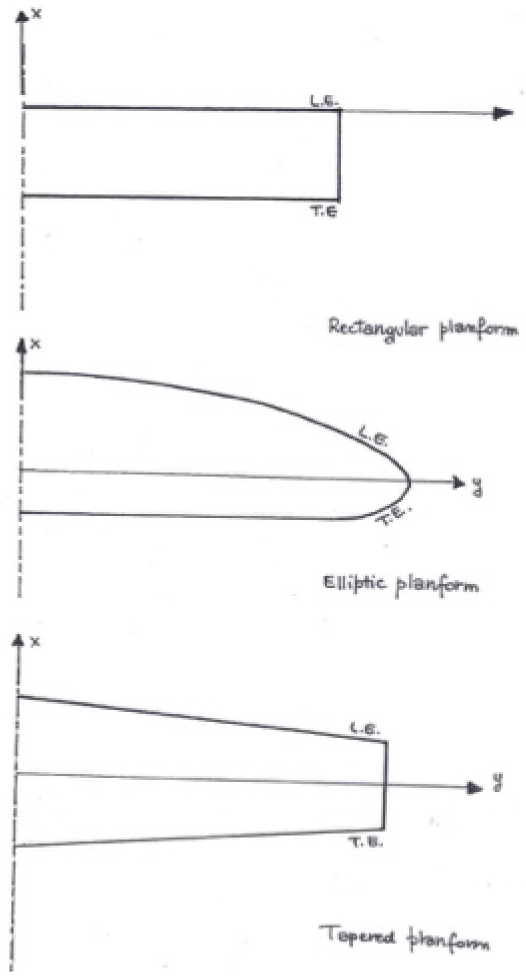


Figure 152: Wing planform main options

In order to select a suitable wing planform and so, to hypothesize a proper value of taper ratio, the most useful and simple approach is to evaluate the variations in terms of lifting capabilities of a family of wing geometries having the same airfoil, and equal geometrical features except for the wing taper ratio. This approach can be carried out in conceptual design phase, exploiting the so called lifting-line theory proposed by Prandtl. With the same approach, it is also possible to evaluate the effect of aspect ratios and wing surface on the lift distribution. Figure 153, Figure 154, Figure 155 and Figure 156 are intended to provide some useful examples of impact of these geometrical characteristics on the lifting distribution.

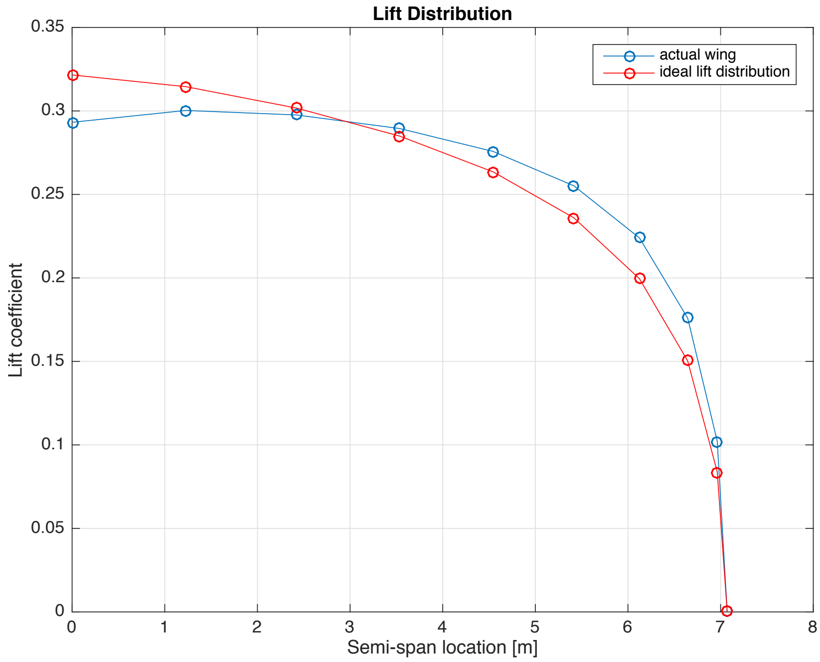


Figure 153: Actual wing lift distribution vs Ideal lift distribution

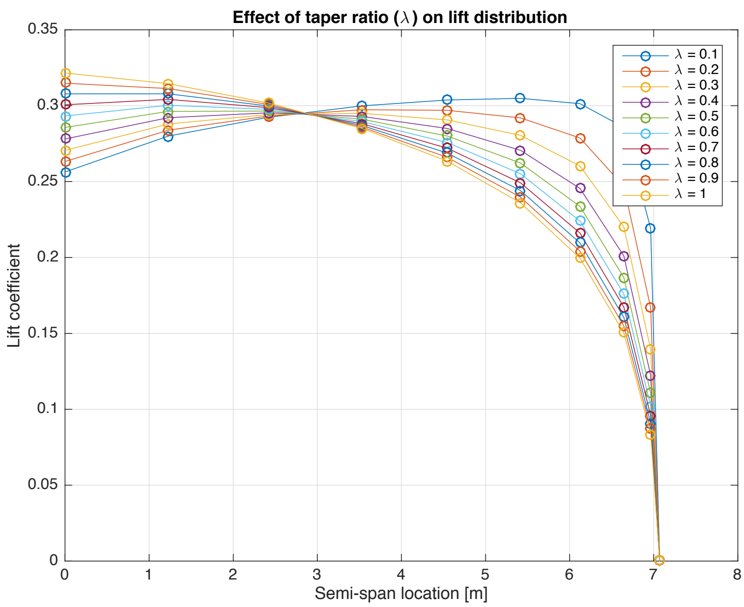


Figure 154: Effect of taper ration on wing lift distribution

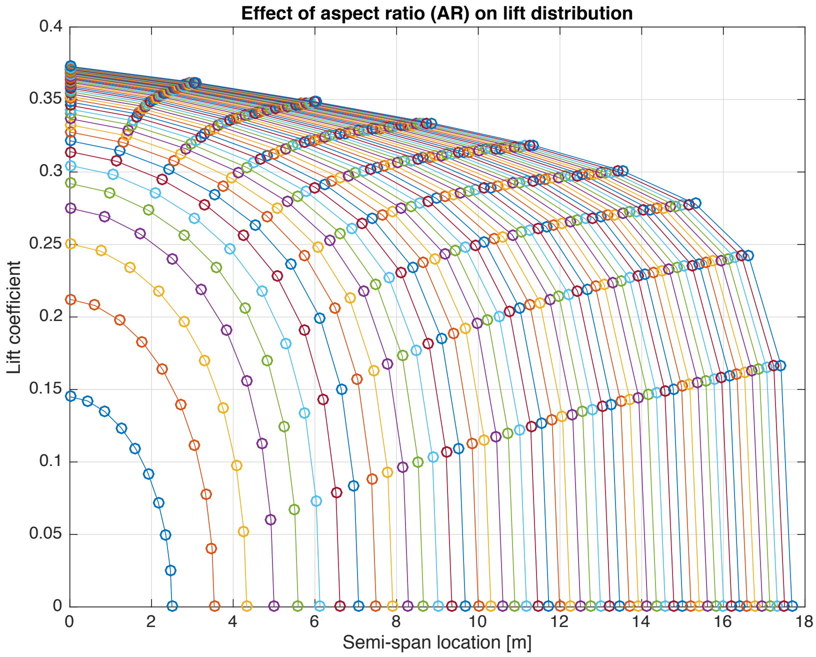


Figure 155: Effect of aspect ratio on wing lift distribution

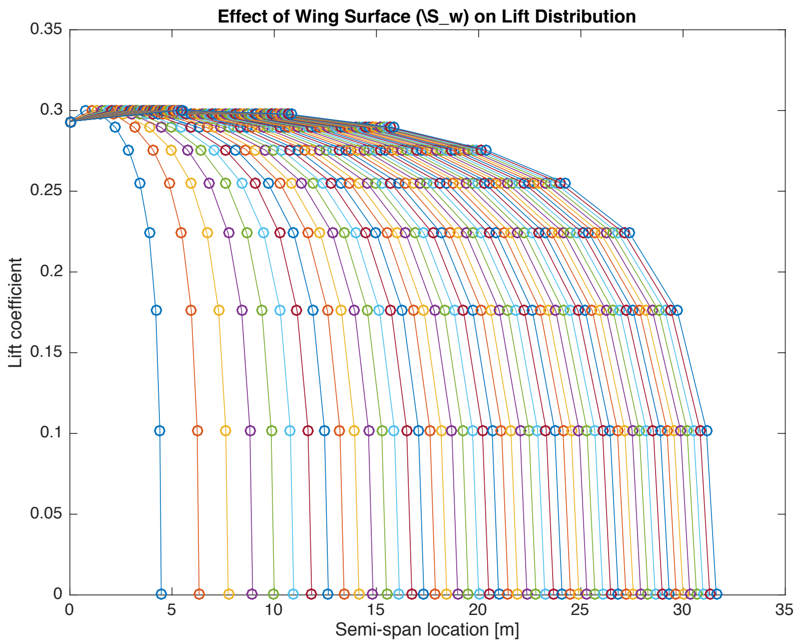


Figure 156: Effect of wing surface on wing lift distribution

Table 82: Impact of requirements on wing design parameters

	List of requirements	Wing Vertical location	Airfoil characteristics			Wing geometry characteristics				
			t/c	LE radius	camber	wing incidence	Aspect Ratio	Sweep Angle	Dihedral Angle	Taper Ratio
Aerodynamic	The wing shall maximize the lift generation		✓			✓	✓	✓	✓	✓
	The wing lifting performances shall be maximized during the cruise phase		✓		✓	✓				
	The wing contribution to the overall drag shall be minimized during the cruise phase		✓	✓	✓	✓				
	During cruise, the drag generated by the fuselage shall be minimized, i.e, the fuselage angle of attack in cruise shall be null.					✓				
	The wing geometry shall minimize the 3D effect due to wing tip vortex.						✓			
	The wing stall shall be avoided			✓			✓			
	The wing shall be able to maximize the lift-over-drag ratio.						✓			
	The wing area shall be included within the Mach cone to withstand the structural and heating loads							✓		
	The wing planform shall maximize the lift distribution.									✓
	The aircraft frontal section shall be minimized.	✓								
	The ground effect shall minimized/maximized.	✓								
	The aircraft configuration shall minimize the overall drag.	✓								
	The aircraft configuration shall maximize the lifting capabilities	✓								
	The aircraft lifting capabilities during take-off and landing operations shall be maximized.			✓						
	The wing configuration shall prevent from bow shock formation			✓						
	The aircraft lower surface shall be as flat as possible, preventing from aero-thermodynamic issues				✓					
	The wing shall be able to generate a TBD amount of lift at zero angle of attack.				✓					
Operations	The available excursion of angle of attack during take-off operation shall be maximized					✓				
	During cruise phase, the fuselage angle of attack in cruise shall be null in order to guarantee the maximum comfort level.					✓				
	The aircraft landing distance shall be minimized.					✓				
	The wing architecture shall guarantee proper clearance to ground and water.								✓	
	The landing gear length shall be minimized	✓								
	The aircraft configuration shall facilitate the loading and unloading operations	✓								
	The aircraft configuration shall guarantee the proper on-ground clearance to the ground	✓								
	The aircraft configuration shall facilitate the refuelling operations	✓								
	The aircraft configuration shall enhance STOL capabilities	✓								
	The aircraft shall be operated from un-prepared fields	✓								
	The aircraft configuration shall guarantee a proper pilot visibility	✓								
	The aircraft configuration shall allow proper floating capabilities in case of emergency landing on waters.	✓								
	The aircraft should be able to operate at high angle of attacks.			✓						
Stability and Control	The wing geometry shall maximize the effectiveness of wing control						✓			
	Lateral stability shall be enhanced.							✓	✓	✓
	Lateral manoeuvrability shall be enhanced							✓		
	The aircraft controllability in turbulence shall be enhanced.									✓
	The aircraft lateral control shall be improved									
	The aircraft pitching moment shall me maximized				✓					
Safety	Gliding performances shall be maximized.						✓			
	The wing stall shall be postponed with respect to the canard stall.						✓			
	The wing stall shall be anticipated with respect to the tail stall.						✓			
	The stall speed shall be increased							✓		
Logistic and maintainance	The aircraft shall be easy to be maintained.								✓	
	Logistic on-ground infrastructures required to perform maintenance shall be reduced.								✓	
Costs	The wing manufacturing costs shall be minimized.									✓
Weight and Balance	The wing weight shall be minimized	✓								✓
	The fuselage weight shall be minimized.	✓								
	The landing gear weight shall be minimized	✓								
Payload and Systems	The volume available to host passengers shall be maximized.	✓								
	The volume available to accommodate payload shall be maximized	✓								
	The volume available to accommodate payload shall be maximized		✓							
Structure	The number of cuts in the fuselage structure shall be reduced	✓								

Table 82 is a matrix that summarizes the impact of several requirements to the wing architecture definition.

5.7 Support Tool Chain

In order to support the wing design activity, a proper Matlab® code has been developed. This section will provide the reader with a brief description of the tool, underlying the main benefits, especially in terms of traceability. Moreover, considering the fact that this design step is a mixed of both sketching and sizing activity, proper interfaces between Matlab® code and other development environments such as Solidworks® and Simulink®, as well as with requirements management tools such as IBM Doors® or configuration management such as IBM Rapsody® have been in-depth analysed, providing a complete tool chain to the final user.

Figure 157 provides a graphical view of the envisaged tool chain. As it is possible to be noticed, the user workload has been reduced thanks to the creation of a Graphical User Interface (GUI), that eases the overall process. This GUI has been developed in a Matlab environment with the aim of supporting the user during the overall process. In particular, this GUI allows to:

- Ease the process of problem definition.
- The management of the overall wing design process.
- Ease design iterations.
- Allowing track changes.

The Matlab® code is related to the spreadsheet generated in Excel environment that contains inputs and outputs of the design process. Some of these Excel files are also constituting the database and thus, they are a useful collection of information. The Excel files provide also the link between the Matlab code and the IBM Doors. At this purpose, Excel files are used to create interfaces between Matlab and Doors and vice versa.

The developed Matlab® code implements the overall approach previously described. In particular, the user, interacting with the GUI, performing the first selections, such as the type of mission required, the role and the maximum achievable mach number. In this case, the user is simply doing selections on the screen but these are precious information to start the overall design process. In particular, thanks to these high level choices, the tool is able to generate a high list of requirements, belonging to different categories, from aerodynamic to operation, from safety to maintenance, simply automatically selecting the most impacting ones from the main matrix in Table 82.

Once the major inputs have been inserted, the Matlab® tool is able to provide the user with the suggestion of the most convenient wing vertical location with respect to the inserted inputs. In particular, the tool will provide the user with a series of ordered sheets, each one presenting a pictorial view of the vertical location and the related list of pros and cons. Considering the crucial role of this selection, the user can decide to accept the suggestion of the tool and proceed in the wing design process with the first ranked configuration. Otherwise, the user can navigate through the other options and select a different one, accepting related pros and cons. This degree of freedom is required because this tool-chain is not intended to force the designer to a frozen solution but supporting in a rational way the creative process of aircraft design.

Then, once the vertical location of the wing with respect to the fuselage has been fixed, the user shall insert some numerical high level estimations that are closely related to what has been done in previous steps, when the aircraft configuration has been selected and the first numerical estimation have been carried out. In this way, the tool can suggest a proper airfoil (or a family of airfoil) suitable for the envisaged application. Also in this case, the designer is not forced to use the suggested airfoil but he/she can decide to move to the next step of the design process directly importing the geometry of the other existing or ad-hoc developed airfoil and the some aerodynamic and geometrical characteristics.

At this stage it is possible to go on with the definition of the optimal geometry for the wing. The results of these evaluations can be accessed by the users in several ways. First of all, a new process of requirements refinement/generation starts, providing an updated list of requirements, properly stored. Then, a proper routine provides the designer with a wing sketch. Moreover, the same data are used to update a 3D parametric CAD model. Using a proper interface between the code (in Matlab environment) and the 3D model, the user can also add some changes in the parametric model and these changes have a direct impact on the requirements. In this way, there is complete traceability between model and requirements. Moreover, the 3D model can be exported to be used in other higher fidelity tools, to perform more detailed analyses such as the aerodynamic and structural ones. In particular, the possibility of importing the CAD model in Simulink® exploiting the SimScape® library that allows to simulate the way of working of the imported 3D components. In particular, this tools connection demonstrated to be very useful in order to test and solve some issues related to the integration of components and equipment within a system. In the case of wing, the simulation of the actuation of movable surfaces or the retraction and extraction of landing gear, can be directly simulated. Like in the case of Solidworks®, also for the Simulink® model there is the possibility of connecting each element or variable to one or more requirements. In case of requirements containing numerical information, there would also be the possibility of verifying them directly during the simulation.

This tool chain has been envisaged at first and here described thinking to the specific case of supporting an aircraft wing design. However, it is crystal clear that this is a general approach that could be implemented for all the other different design areas. Moreover, the possibility of maintaining the traceability of the overall process shows is major benefits with respect to the traditional approach, in case of complex systems. It can be useful noticing that in the example of requirements implementation in DOORS® (Figure 159) there is the possibility of relating the impact of each requirement onto the basic wing design parameters, with the possibility of reflecting all the theoretical investigations reported above.

Furthermore, as it is illustrated in Figure 157, the introduction of a Flight Simulator, like X-Plane, has been envisaged in order to test and verify additional characteristics such as the handling qualities or different flying performances, that are hardly quantifiable at this high level of design.

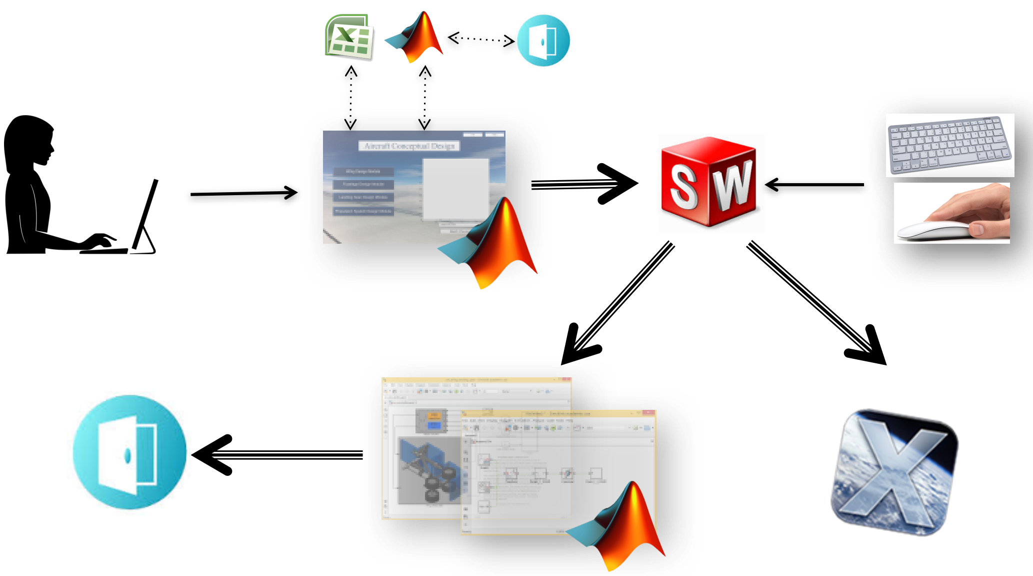


Figure 157: Pictorial view of the envisaged tool-chain for wing design

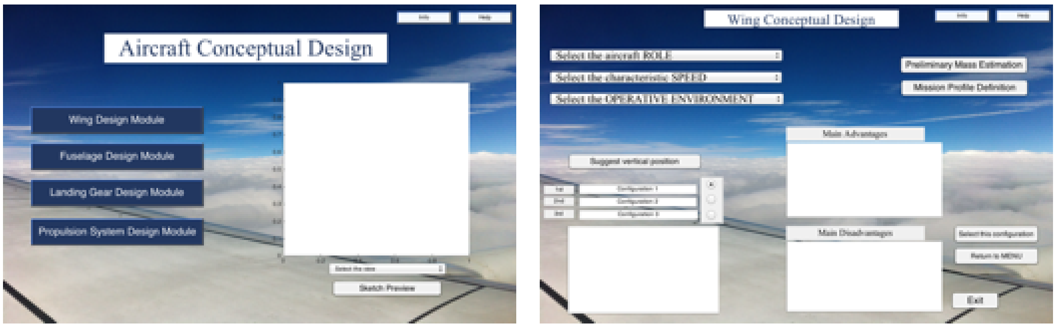


Figure 158: Ad-hoc Matlab® GUI developed to support the wing conceptual design

ID		Wing Vertical Location	Airfoil	Wing Geometry
R51	**1 High Level Requirements**	N/A	N/A	N/A
R50	**1.1 Aerodynamic Requirements**	N/A	N/A	N/A
R1	The wing shall maximize the lift generation	No	t/c	ALL
R2	The wing lifting performances shall be maximized during the cruise phase	No	t/c & Camber	Wing incidence
R3	The wing contribution to the overall drag shall be minimized during the cruise phase	No	ALL	Wing incidence
R4	During cruise the drag generated by the fuselage shall be minimized i.e the fuselage angle of attack in cruise shall be null.	No	None	Wing incidence
R5	The wing geometry shall minimize the 3D effect due to wing tip vortex.	No	None	Aspect Ratio
R6	The wing stall shall be avoided	No	LE Radius	Aspect Ratio
R7	The wing shall be able to maximize the lift-over-drag ratio.	No	None	Aspect Ratio
R8	The wing area shall be included within the Mach cone to withstand the structural and heating loads	No	None	Sweep Angle
R9	The wing planform shall maximize the lift distribution.	No	None	Taper Ratio
R10	The aircraft frontal section shall be minimized.	Yes	None	None
R11	The ground effect shall minimized/maximized.	Yes	None	None
R12	The aircraft configuration shall minimize the overall drag.	Yes	None	None
R13	The aircraft configuration shall maximize the lifting capabilities	Yes	None	None
R14	The aircraft lifting capabilities during take-off and landing operations shall be maximized.	No	LE Radius	None
R15	The wing configuration shall prevent from bow shock formation	No	LE Radius	None
R16	The aircraft lower surface shall be as flat as possible preventing from aero-thermodynamic issues	No	Camber	None
R17	The wing shall be able to generate a TBD amount of lift at zero angle of attack	No	Camber	None

Figure 159: Example of Wing Requirements implementation on DOORS®

5.8 Application to the design of a wing for a suborbital vehicle

This and the following paragraph aim at demonstrating that the applicability of the presented methodology to the design of highly competitive transportation systems. In particular, this section collects the result of the application of the wing design methodology to a suborbital vehicle.

The final aircraft should be able to perform a suborbital flight allowing two passengers to perform space tourism activities, such as experiencing microgravity and observing the Earth curvature from 100 km of altitude.

Looking at the requirements matrix, previously presented, only a subset of requirements can be usefully applied to this case study, taking into account the stakeholder analysis and the mission analysis.

Table 83: Selection of impact of requirements on wing design parameters

	Suborbital	List of requirements	Wing Vertical location	Airfoil characteristics			Wing geometry characteristics				
				t/c	LE radius	camber	wing incidence	Aspect Ratio	Sweep Angle	Dihedral Angle	Taper Ratio
Aerodynamic	■	The wing shall maximize the lift generation		✓			✓	✓	✓	✓	✓
		The wing lifting performances shall be maximized during the cruise phase									
		The wing contribution to the overall drag shall be minimized during the cruise phase									
		During cruise, the drag generated by the fuselage shall be minimized, i.e, the fuselage angle of attack in cruise shall be null.									
	■	The wing geometry shall minimize the 3D effect due to wing tip vortex.						✓			
	■	The wing stall shall be avoided			✓			✓			
	■	The wing shall be able to maximize the lift-over-drag ratio.						✓			
	■	The wing area shall be included within the Mach cone to withstand the structural and heating loads							✓		
	■	The wing planform shall maximize the lift distribution.									✓
		The aircraft frontal section shall be minimized.									
	■	The ground effect shall minimized/maximized.	✓								
	■	The aircraft configuration shall minimize the overall drag.	✓								
	■	The aircraft configuration shall maximize the lifting capabilities	✓								
	■	The aircraft lifting capabilities during take-off and landing operations shall be maximized.			✓						
	■	The wing configuration shall prevent from bow shock formation			✓						
	■	The aircraft lower surface shall be as flat as possible, preventing from aero-thermodynamic issues				✓					
		The wing shall be able to generate a TBD amount of lift at zero angle of attack.									
Operations		The available excursion of angle of attack during take-off operation shall be maximized.									
		During cruise phase, the fuselage angle of attack in cruise shall be null in order to guarantee the maximum comfort level.									
		The aircraft landing distance shall be minimized.									
	■	The wing architecture shall guarantee proper clearance to ground and water.								✓	
	■	The landing gear length shall be minimized	✓								
	■	The aircraft configuration shall facilitate the loading and unloading operations	✓								
	■	The aircraft configuration shall guarantee the proper on-ground clearance to the ground	✓								
	■	The aircraft configuration shall facilitate the refuelling operations	✓								
		The aircraft configuration shall enhance STOL capabilities									
		The aircraft shall be operated from un-prepared fields									
	■	The aircraft configuration shall guarantee a proper pilot visibility	✓								
	■	The aircraft configuration shall allow proper floating capabilities in case of emergency landing on waters.	✓								
	■	The aircraft should be able to operate at high angle of attacks.			✓						
Stability and Control	■	The wing geometry shall maximize the effectiveness of wing control						✓			
	■	Lateral stability shall be enhanced.							✓	✓	✓
	■	Lateral manoeuvrability shall be enhanced							✓		
		The aircraft controllability in turbulence shall be enhanced.									
	■	The aircraft lateral control shall be improved									
		The aircraft pitching moment shall me maximized									
Safety	■	Gliding performances shall be maximized.						✓			
		The wing stall shall be postponed with respect to the canard stall.									
	■	The wing stall shall be anticipated with respect to the tail stall.						✓			
	■	The stall speed shall be increased							✓		
Logistic and maintainance	■	The aircraft shall be easy to be maintained.								✓	
		Logistic on-ground infrastructures required to perform maintenance shall be reduced.									
Costs		The wing manufacturing costs shall be minimized.									
Weight and Balance	■	The wing weight shall be minimized	✓								✓
	■	The fuselage weight shall be minimized.	✓								
	■	The landing gear weight shall be minimized	✓								
Payload and Systems	■	The volume available to host passengers shall be maximized.	✓								
		The volume available to accommodate payload shall be maximized									
	■	The volume available to accommodate systems shall be maximized		✓							
Structure	■	The number of cuts in the fuselage structure shall be reduced	✓								

5.8.1 Wing Vertical Location

Looking at the matrix in Table 83, it is easy noticing that the wing vertical location for this configuration is affected by requirements and constraints belonging to different areas of interest: aerodynamic, operations, weight and balance, payload and systems and structure.

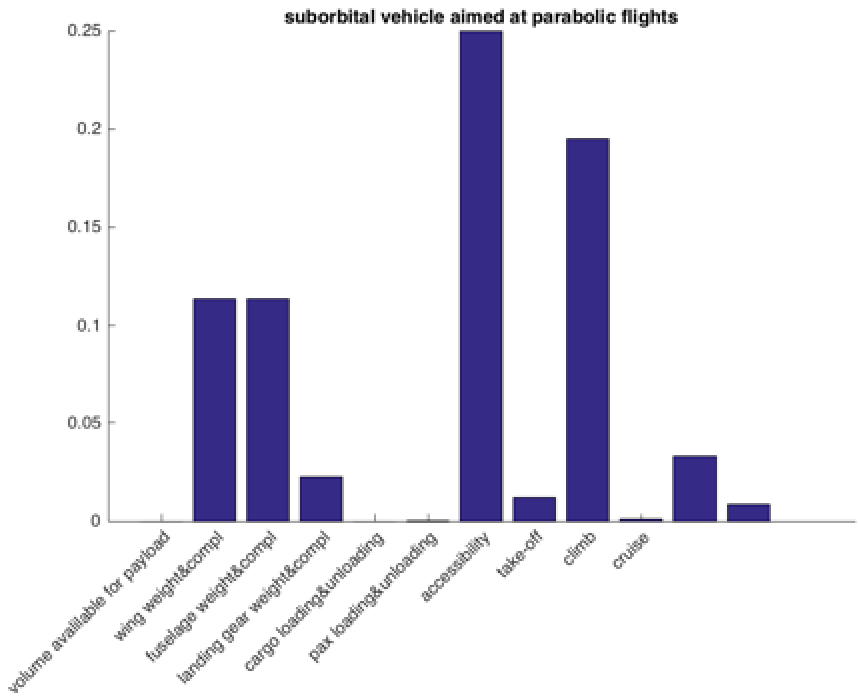


Figure 160: Different weighting factors for the identified areas of interest

The application of the methodology shows that for this kind of application, the most suitable wing vertical location with respect to the fuselage, is the low one. This appears to be a reasonable result thinking that this kind of transportation system should be able to perform a re-entry part of mission profile (besides it is not a re-entry from an orbital altitude) and the floating capability required in case of emergency landing on water. However, as indicated in previous reported Figure, direct output of the ad-hoc developed Matlab® tool, the low wing vertical location has some relevant drawbacks that could be furtherly overcome properly playing with different geometrical parameters.

	WeightingFactors	LOW 1	LOW 2	MID 1	MID 2	HIGH 1	HIGH 2	HIGH 3	HIGH 4
WingWeight&Complexity	0	8	10	3	6	8	8	10	10
FuselageWeight&Complexity	0.1136	7	8	7	7	7	7	10	10
LandingGearWeight&Complexity	0.1136	8	9	6	5	5	5	6	6
CargoLoading&Unloading	0.0227	9	9	7	7	7	7	7	7
PaxLoading&Unloading	4.2228e-10	8	8	4	4	7	7	7	7
Accessibility	2.9683e-08	8	8	4	6	10	10	10	10
TakeoffAerodyn&Hq	0.2500	8	8	5	5	8	8	8	8
ClimbAerodyn&Hq	0.0122	6	6	8	8	7	7	7	7
CruiseAerodyn&Hq	0.1951	8	8	10	10	4	4	4	4
LandingAerodyn&Hq	0.0013	8	8	10	10	5	8	6	6
RESULTS	0.0329	10	10	7	7	7	5	3	5
	0.0085	4	4	8	8	10	10	10	10
	0.0414	10.0341	10.0341	7.0683	7.0683	7.0853	5.0853	3.0853	5.0853

Figure 161: Example of output coming from the Matlab GUI with the results of the trade-off for the optimal wing vertical location

5.8.2 Wing Airfoil definition

Considering the wing airfoil definition, before selecting a proper airfoil, it is important to have an idea of the leading edge radius, camber and thickness that can be selected.

Following the methodology described in the previous paragraphs and thanks to the support of the ad-hoc built-in Matlab® tool, the main airfoil parameters have been estimated. In particular, the following table summarizes the results obtained for this case study, with relative comments about the proposed solutions.

Table 84: Selection of airfoil characteristics for the reference case study

Characteristic		**Comments**
Leading Edge radius	Large-to-intermediate	Considering the specific mission profile, the most important requirement affecting this selection is the need of guaranteeing the capability of flying and performing maneuvers at high angles of attack.
Camber	Double cambered	This solution allows the airfoil to guarantee a certain amount of lift. This is extremely useful in this case in which the aircraft should be able to perform a vertical take-off. The lower surface will be only moderated cambered in order to withstand to the aero-thermo-dynamic loads.
Thickness	$(t/c)_{max}<0.09$	A thin airfoil has been proposed taking into account the speed regime that the

aircraft shall guarantee. However, considering the range of numerical values proposed for the supersonic speed regime, the highest estimation has been considered in order to partially satisfy the need of free room to install systems within the wing.

Then, it should be necessary to find out if an existing airfoil could be selected for this application. Considering the peculiarities in terms of wide speed and altitude ranges, it is convenient to look at some existing ad-hoc developed airfoil for similar applications and verify that the aerodynamic characteristics could match the designer expectations. In particular, an airfoil similar to the designed for the Space Shuttle can be exploited (Hirschel, 2009). Considering the difference in terms of maximum speed, the analysis of the lifting coefficient variations are here limited to the speed range of interest.

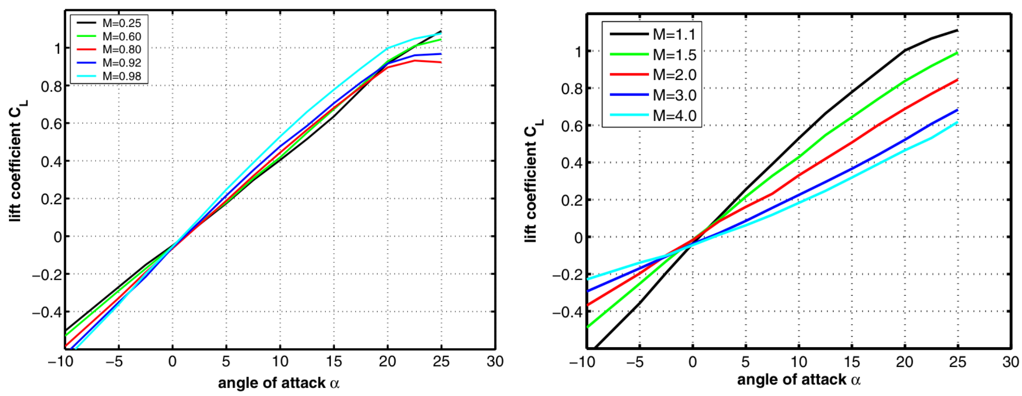


Figure 162: Example of lift coefficient trends at different Mach numbers for a simple reference airfoil selected in conceptual design.

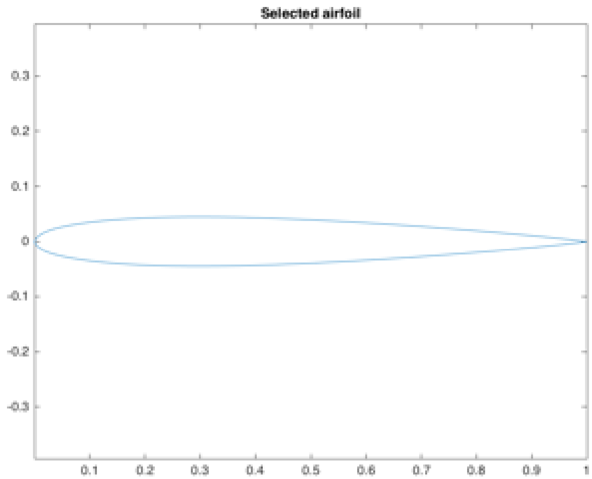


Figure 163: Simple graphical representation of airfoil in the Matlab® GUI

5.8.3 Wing geometry definition

Once the 2D airfoil has been selected, it is possible to finalize the 3D wing design defining all the other geometrical characteristics of the wing.

In the following table, the major results are reported.

Table 85: Selection of wing characteristics for the reference case study

Characteristic		Comments
Wing incidence	α_{set} = 1 deg (fixed)	A fixed incidence will be adopted in order to avoid higher maintenance costs and increasing risk. The numerical value is the results of the application of the statistical approach whose outcome are summarized in table #.
Aspect Ratio	AR = 3	The selected aspect ratio is relatively low considering the typical aeronautical scenario. However, it is perfectly compliant with the AR values of existing suborbital vehicles. In particular, besides

		the fact that this choice may not be the optimal one from the aerodynamic point of view, it has several other benefits. Indeed, as far as stability and control is concerned, this value moves away the risk of aileron reversal. Moreover, the CG shift due to the fuel consumption results to be reduced.
Wing Sweep Angle	Λ = 79.7 deg (fixed)	Considering that the envisaged mission profile has not so wide speed ranges to be faced with, because the aircraft will not reach hypersonic Mach numbers, a single wing sweep strategy can be suitable. The numerical values obtained by the estimation guarantees the overall wing surface to stay within the Mach cone.
Dihedral Angle	Γ = 1; (positive)	A small positive dihedral angle is suggested to take into account the low wing selected configuration and the supersonic flight regime, enhancing the lateral stability and the on ground clearance. However, higher values cannot be adopted to allow vertical take-off in not tail-sitting position.
Taper Ratio	λ = 0.15 (quasi Delta wing)	Delta wing configuration provides the aircraft optimal lateral control and spiral stability, allowing a weight reduction, due to an optimized material distribution. However, as it shown in Figure #, this solution is not providing the designer with the best lifting distribution. This problem is here accepted considering that a proper design of the fuselage and of the interface between wing and fuselage can be properly pursued in next design steps.

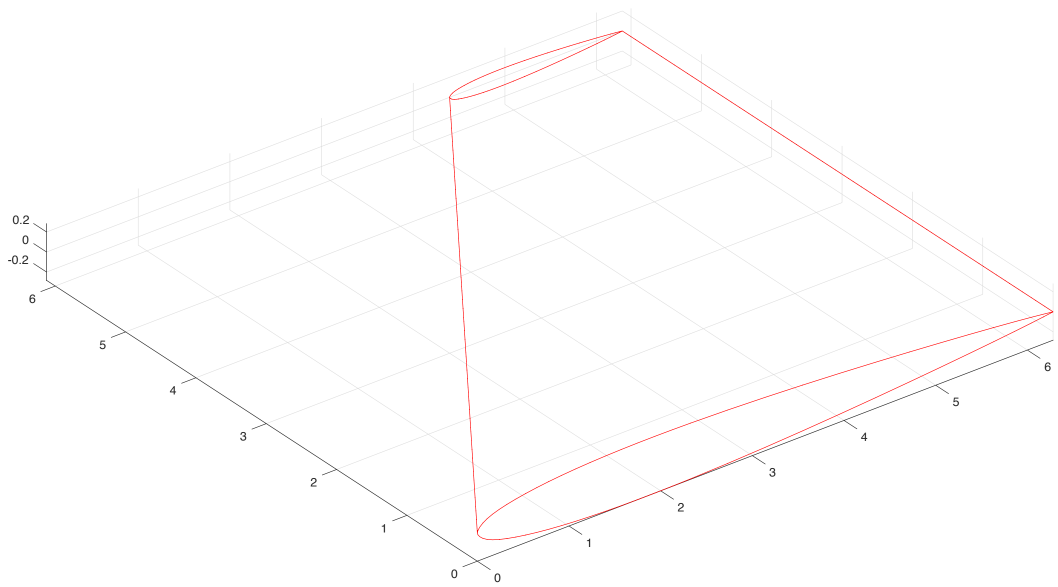


Figure 164: Simple graphical representation of the under-development 3D wing in the Matlab® GUI

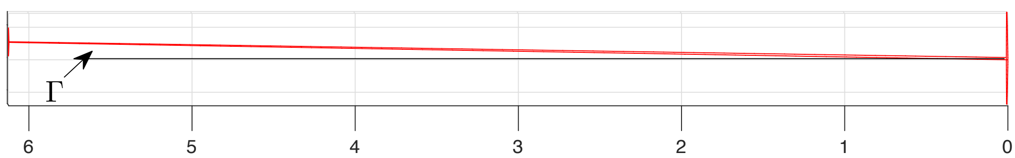


Figure 165: Front view of under-development 3D wing in the Matlab® GUI

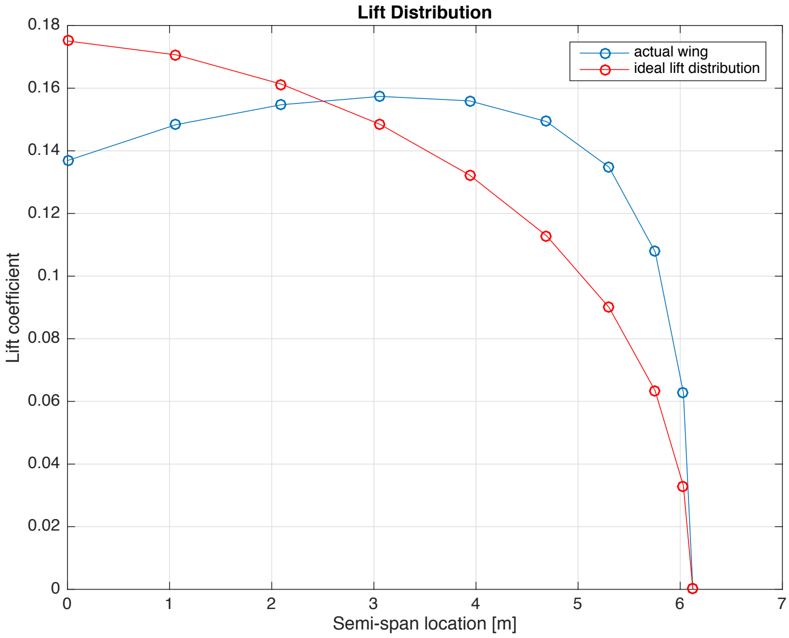


Figure 166: Lift distribution for the case-study wing

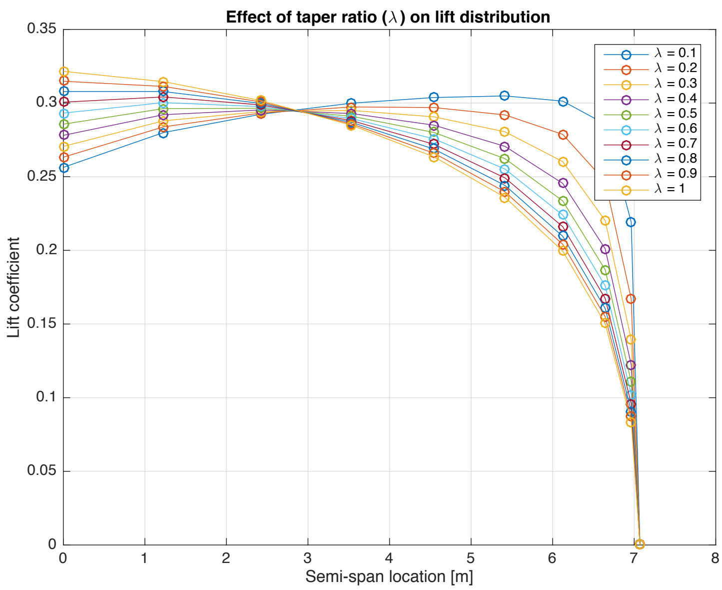


Figure 167: Effect of taper ratio on lift distribution for the case-study wing

It is clear that this is only the very first step in the definition of a wing, especially for hypersonic vehicles. However, it is the fundamental step toward further investigations, in the different specialist disciplines. Furthermore, the presented tool chain, consisting of both commercial and under development tool provides very useful output both in terms of requirements and models to be easily imported in other specific domain software.

Eventually, it has to be noticed that neither control surfaces nor wing internal structure have been dealt with in this chapter because the author decided to insert them within Chapter 7, aims at describing the complex activity of systems integration within the airframe.

References

Hirschel, E. H., & Weiland, C. (2009). Selected Aerothermodynamic Design Problems of Hypersonic Flight Vehicles, *vol. 229. Progress in Astronautics and Aeronautics*, AIAA, Reston, VA, 371-373.

Raymer, D. P., (2012). Aircraft design: A conceptual approach, *American Institute of Aeronautics and Astronautics*. Inc., Reston, VA (2012): 21.

Sadraey, Mohammad H. (2012) Aircraft design: A systems engineering approach. *John Wiley & Sons,* 2012.

Chapter 6

Fuselage Design

6.1 Introduction to fuselage design

The design of the fuselage is of absolute relevance in the framework of Aircraft Design. Besides the fact that its well-known main purpose is to properly accommodate payload, in this chapter, the impact of other requirements on the design of the fuselage is in-depth analysed. In general, the reader will notice that the fuselage design process for the specific case of a hypersonic vehicle does not differ so much from the innovative approaches currently applied for the procedures adopted in aircraft design. However, differently from what is currently proposed in existing literature, the algorithm here proposed is completely integrated within the vehicle design methodology, fully formalized following SysML language, guaranteeing a complete internal and external traceability with the aim of easing the iterative and recursive processes.

Furthermore, also in this case, the absence of any precise laws or certification specifications regulating the design activities for the case of hypersonic vehicles, pushes the author to suggest a hybrid exploitation of regulations, considering those currently used in both aviation and space framework as guidelines. Especially for the emergency provisions and furniture, CS 25 has been considered as a reference document.

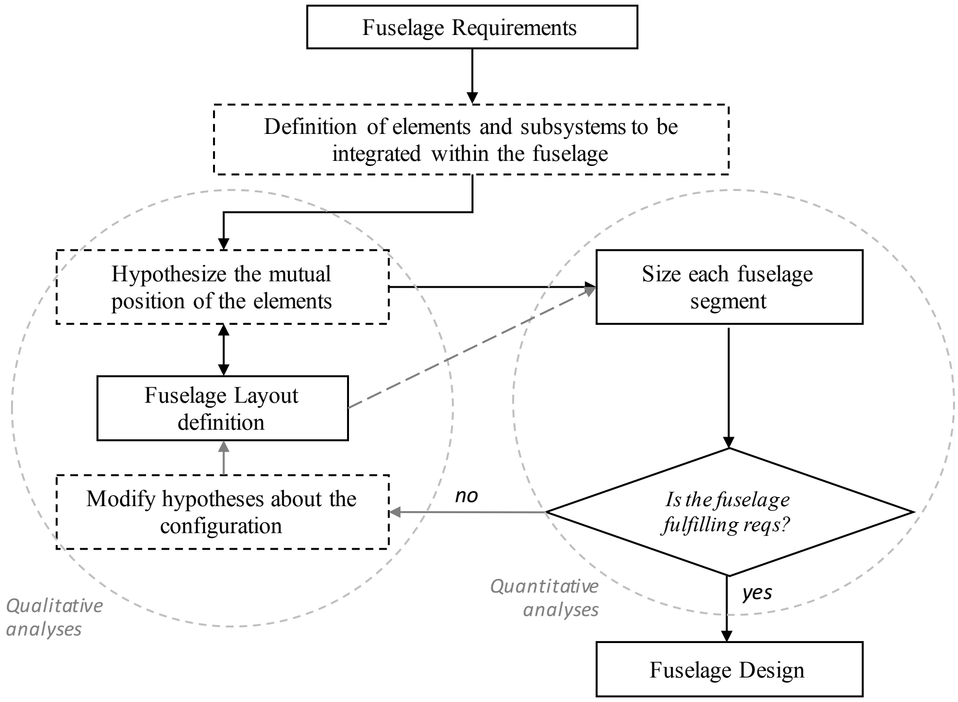


Figure 168: Fuselage design process overview

From a more technical point of view, it has to be noticed that the fuselage design, as well as all the other high level design activities, shall be a balanced mixed of qualitative and quantitative analyses, continuously facing with the challenging integration of different subsystems. It is mainly for this reason that at the beginning of the process, when the general fuselage layout shall be selected, the designer has not got all the elements to carry out a proper trade-off and some selected configurations, after a first sizing attempt, may result to be unfeasible or anyway not optimized for the declared purposes. A clear example could be the case in which a designer has already sized and modelled the crew and passengers' compartment and trying to fit the propulsion subsystem in the fuselage, it becomes clear that the diameter of the overall propulsive group is much larger (or narrow) than the already sized diameter for the front part. For this reason, the reader shall consider that the sketches, reported while presenting possible layout configurations, are merely qualitative and might be not in scale. For this reason, at the first stage, the selection of a configuration can only be hypothesized but the actual choice can be refined and confirmed after proper numerical evaluations only. Thus, after the qualitative design phase, useful mathematical algorithms are suggested to carry out estimations for the different fuselage compartments. Figure 168 tries to summaries the overall activities flow highlighting the major connections between inputs, hypotheses and the main design outcomes (both qualitative and quantitative ones). In addition, periodical requirements checks are suggested as well as the need of iterations aimed at identifying a feasible and realistic configuration.

6.2 Fuselage functionalities and system-level requirements

In this second section of the Chapter focusing on fuselage design, the major objectives and requirements with a specific impact on fuselage design are listed and commented. In addition, a categorization depending on the different Areas of Interest is used. With the aim of following a Systems Engineering approach, the design of the fuselage shall start with the identification of functions that could be exploited by this part of the vehicle. It is clear that, the list of functionalities varies accordingly to each specific design case and for this reason, in this context, all the possible functions have been derived focusing only on the type of applications, i.e. hypersonic vehicles. Starting from the primary function that has already been mentioned above, i.e. to safely host payload (manned and unmanned), a list of additional and more detailed functions that could be performed by the fuselage has been derived.

- To accommodate crew members
- To accommodate passengers
- To accommodate flight attendants and other technical personnel
- To host landing gear
- To host propulsion subsystem
- To host propellant subsystem
- To host avionic subsystem
- To host Environmental Control and Life Support subsystem
- To host additional on-board subsystem
- To provide arm for empennages
- To integrate a Cabin Escape System.

As the reader can notice, the function "to ensure survivability in case of emergency" has not been included yet in this list because all the functionalities required to guarantee the survivability of humans on-board in case of catastrophic failures, will be directly derived by the higher level function that has been allocated to the Escape System However, it is clear that the CES will be a detachable part of the fuselage and for this reason, the function “to integrate a Cabin Escape System” has been considered.

Rigorously following a systems engineering approach, a behavioural perspective shall be analysed. However, with the purpose of design and size the fuselage, its behavioural has not been considered in this chapter. Conversely, the allocation of the different functions to the various fuselage sections and components have been performed (Table 86). This will allow the designer to understand the relative impact of requirements along the designing and sizing process. In this case, the Figure below, does not represent an orthodox function-device matrix, but it is mainly used as guide to trace the impact of functions to be achieved and the design of the several fuselage sections.

However, exploiting SysML language, this matrix can be formalized indicating that the connections between fuselage sections and functions are “allocations” whose specific meaning in this case is only to suggest possible functions allocations on devices (Figure 169 and Figure 170).

Table 86: Functions allocation on the different fuselage sections

	Fuselage					
	Nose	**Cockpit**	**Crew Compartment**	**Cabin Compartment**	**Systems Compartment**	**Tail cone**
To accommodate crew members						
To accomodate passengers						
To accommodate flight attendants and other technical personnel						
To host landing gear						
To host propulsion subsystem						
To host propellant subsystem						
To host avionic subsystem						
To host Environmental Controland LifeSupport subsystem						
To host additional on-board subsystem						
To provide arm for empennages						
To integrate a Cabin Escape System.						

	Nose	Cockpit	Crew_compartment	Cabin_compartment	Systems_compartment	Tail_cone
To_accomodate_crew_members			Crew_compartment			
To_accomodate_passengers				Cabin_compartment		
To_accomodate_flight_attendants_and_other_technical_personnel			Crew_compartment			
To_host_landing_gear					Systems_compartment	
To_host_propulsion_subsystem					Systems_compartment	
To_host_propellant_subsystem					Systems_compartment	
To_host_avionic_subsystem	Nose	Cockpit				
To_host_Environmental_Control_and_Life_Support_System			Crew_compartment	Cabin_compartment	Systems_compartment	
To_host_additional_on_board_subsystems					Systems_compartment	
To_provide_arm_for_empennages					Systems_compartment	Tail_cone
To_integrate_a_Cabin_Escape_System		Cockpit	Crew_compartment	Cabin_compartment	Systems_compartment	

Figure 169: Functions allocation on the different fuselage sections (MBSE)

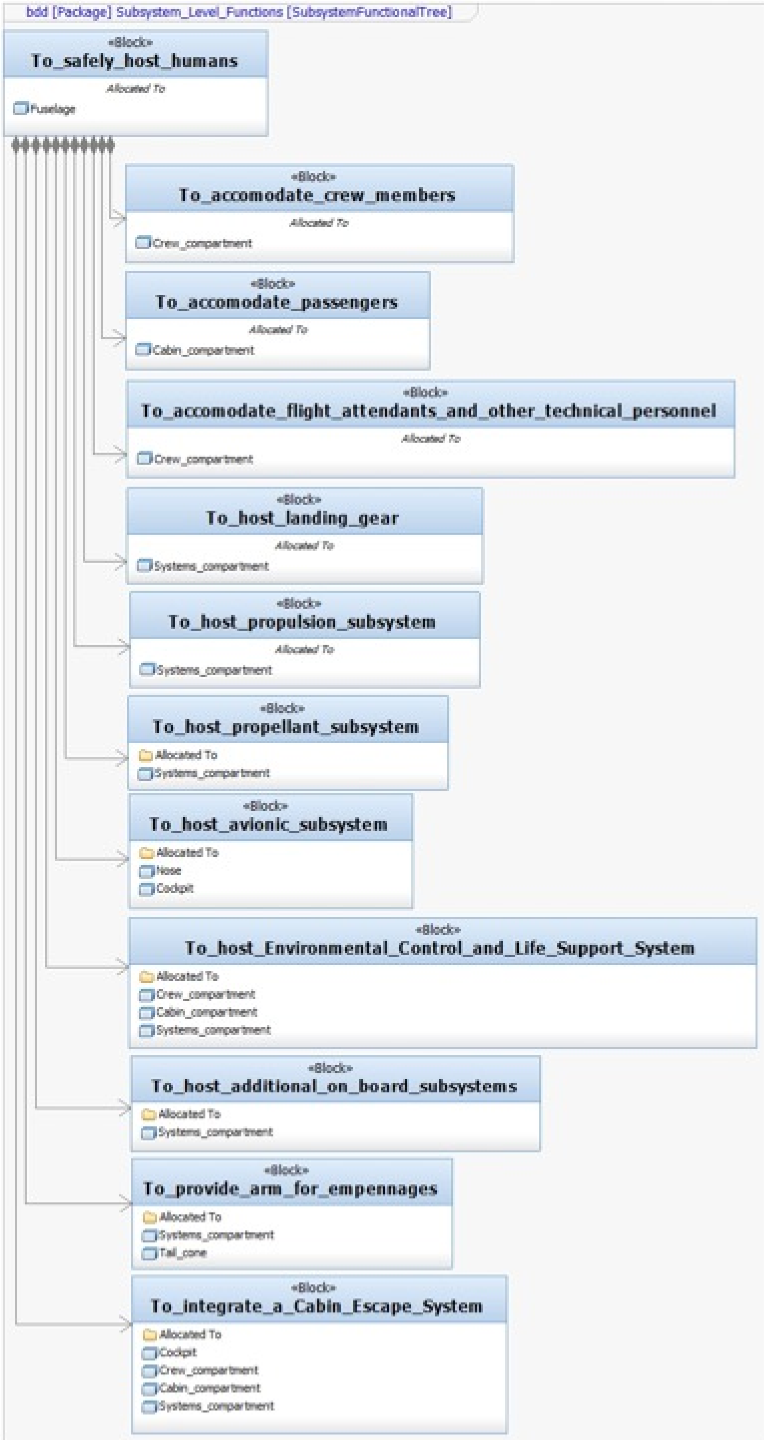


Figure 170: Functional Tree at subsystem level

From this list of functions to be achieved by the fuselage, a first list of requirements (mainly functional and lower-level mission requirements) at system level can be derived, allowing to start the design activities. Please, notice that the list of requirements is not only a mere transcription or modification of the functions list but it contains also the formalization of some evident or hidden stakeholders' desires collected through a proper analysis. Moreover, following the same approach used for wing design, the requirements have been categorized depending on some Areas of Interest:

Comfort and safety:

- The fuselage shall safely accommodate crew members.
- The fuselage shall safely accommodate crew passengers.
- The fuselage shall guarantee proper room allowing passengers to experience microgravity
- The fuselage shall guarantee proper room allowing scientists to carry out experiments.
- The fuselage shall guarantee a proper view of the Earth.

Aerodynamic:

- The fuselage shall generate the lowest possible drag.
- The fuselage shall contribute positively to the lift generation
- The fuselage wetted area shall be minimized.

Structure:

- The fuselage weight shall be minimized.
- The fuselage shall sustain the structural loads all along the flight profile.
- The fuselage shape shall be as symmetric as possible.
- The fuselage shall accommodate landing gear.
- The fuselage shall accommodate propulsion subsystems.
- The fuselage shall accommodate propellant subsystems.
- The fuselage shall accommodate avionic subsystems
- The internal arrangement shall guarantee the proper centre of gravity location.
- The overall fuselage structure shall be able to separate the cabin escape system.

- The fuselage shall properly accommodate cargo

Logistics and Operations:

- The fuselage shall ease loading and unloading operations
- The fuselage shall guarantee proper airworthiness characteristics.
- The cockpit shall be properly designed in order to allow visibility to the pilots.
- The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.

With the aim of making a comparison between wing and fuselage design, the list of requirements shows that while for the wing, aerodynamic and structural requirements are the most impacting on the overall design process, in the case of the fuselage, several requirements are strictly related to accommodation or safety issues. In particular, in the case of hypersonic or suborbital transportation systems, the need of enhancing the safety levels by means of a detachable escape system, implies the designers to elaborate new fuselage design. With the same approach used for functions, the allocation of requirements on the identified fuselage sections has been carried out (Table 87) and also implemented in a MBSE approach (Figure 171). Then, in the following sections, when the design of the different fuselage section will be investigated, the requirements will also be allocated to the design parameters which they impact on.

6.3 Possible architecture configurations

As a first step in the definition of the most suitable fuselage architecture, it is important to understand which subsystems are intended to be installed within the fuselage and immediately after, in which fuselage sections they might suit, hypothesizing the proper location for the different elements.

Aiming, in particular, at defining the most suitable fuselage layout for a hypersonic transportation system, the following subsystems have been considered to be potentially hosted in fuselage. It is clear that depending on the specific case-study, some subsystems can absolutely be absent in the entire transportation system or present in other parts of the vehicle and simply not installed in fuselage.

Table 87: Requirements allocation on the different fuselage sections

		Fuselage					
		Nose	Cockpit	Crew Compartment	Cabin Compartment	Systems Compartment	Tail cone
Comfort and Safety	*The fuselage shall safely accommodate crew members*						
	The fuselage shall safely accommodate passengers						
	The fuselage shall guarantee proper room allowing passengers to experience microgravity						
	The fuselage shall guarantee proper room allowing scientists to carry out experiments.						
	The fuselage shall guarantee a proper view of the Earth.						
Aerodynamic	*The fuselage shall generate the lowest possible drag.*						
	The fuselage shall contribute positively to the lift generation						
	The fuselage wetted area shall be minimized.						
Structure	*The fuselage weight shall be minimized.*						
	The fuselage shall sustain the structural loads all along the flight profile.						
	The fuselage shape shall be as symmetric as possible.						
	The fuselage shall accommodate landing gear.						
	The fuselage shall accommodate propulsion systems.						
	The fuselage shall accommodate avionic subsystems						
	The fuselage shall accommodate propellant subsystem						
	The internal arrangement shall guarantee the proper centre of gravity location.						
	The overall fuselage structure shall be able to separate the cabin escape system.						
	The fuselage shall properly accommodate cargo						
Logistics and Operations	*The fuselage shall ease loading and unloading operations*						
	The fuselage shall guarantee proper airworthiness characteristics.						
	The cockpit shall be properly designed in order to allow visibility to the pilots.						
	The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.						

	FR28	FR29	FR30	FR31	FR32	FR33	FR34	FR35	FR36	FR37
Nose						FR33	FR34	FR35	FR36	FR37
Cockpit						FR33		FR35	FR36	FR37
Crew_compartment	FR28			FR31		FR33		FR35	FR36	FR37
Cabin_compartment		FR29	FR30	FR31	FR32	FR33		FR35	FR36	FR37
Aft_fuselage						FR33		FR35	FR36	FR37
Tail_cone						FR33		FR35	FR36	FR37

	FR38	FR39	FR40	FR41	FR42	FR43	FR44	FR45	FR46	FR47
Nose	FR38		FR40		FR42	FR43			FR46	FR47
Cockpit	FR38		FR40		FR42	FR43		FR45	FR46	FR47
Crew_compartment	FR38				FR42	FR43		FR45	FR46	
Cabin_compartment	FR38				FR42	FR43	FR44	FR45	FR46	
Aft_fuselage	FR38	FR39		FR41	FR42				FR46	
Tail_cone	FR38				FR42				FR46	

Figure 171: Requirements allocation on the different fuselage sections (MBSE)

This is the list of the subsystems that could have the strongest impact on the fuselage design. Please, notice that most of the subsystems are applicable for different types of aircraft, while some others are specifically enviable for hypersonic transportation systems only.

- Passenger compartment
    - On-board systems mainly related to safety provisions and life support
- Crew compartment
    - Avionic subsystem
- Propulsion subsystems
    - Air-breathing engine inlet
    - Exhaust gas nozzles
- Propellant subsystem
- Additional subsystems usually located in fuselage:
    - Landing gears
    - Wing box
    - Thermal and Energy Management subsystems.

Moreover, the following general design recommendations, guiding the designers in properly outlining the fuselage layout should be followed as well:

1. For visibility reasons, the cockpit shall be located in an advanced position.
2. For safety concerns, the propulsion systems and related tanks, shall be located as far as possible from the crew and passengers compartment. This will also ease the possibility of safely separating the front part of the fuselage, where crew and passengers are hosted and allowing a proper escape system.

Depending on the different capabilities requested to the fuselage, the following configurations, here ranked for increasing complexity, have been considered:

- <u>Conf. 1</u>: Fuselage with a passengers’ compartment only
- <u>Conf. 2</u>: Fuselage with crew and passengers’ compartment
- <u>Conf. 3</u>: Crew and passengers’ compartments + air-breathing engines
- <u>Conf. 4</u>: Crew and passengers’ compartments + rocket motor

- Conf. 5: Crew and passengers' compartments + rocket motor + propellant tanks
- Conf. 6: Crew and passengers' compartments + air-breathing engines+ fuel tanks
- Conf. 7: Crew and passengers' compartments + air-breathing engines +rocket motor
- Conf. 8: Crew and passengers' compartments + air-breathing engines +rocket motor + fuel tanks + propellant tanks

In the following subsections, a synthetic and qualitative description for each of this configuration is provided. In addition, it has to be noticed that especially when dealing with air-breathing propulsion systems, additional requirements mainly related to the capability of performing vertical take-off and landing must be taken into account, even if this may result in configuration changes. Furthermore, the presence and integration of air-intakes may be another challenging issue. However, the fact of considering these requirements mainly related to integration issues, since the beginning of the design process, will prevent the designer coming up with unfeasible solutions or wasting time analysing configurations that would be too far from an optimal condition.

Eventually, it has to be noticed that the selection of the best alternative shall be supported by the first numerical evaluations that would give a first feedback on the feasibility of the project. For this reason, once the major configuration alternatives are presented, Section 6.4 aims at suggesting suitable algorithms for the definition of the first sizing. The algorithms proposed in the following subsections are generic enough to be exploited for the sizing of all the configuration described in this Section.

6.3.1 Conf. 1: Passengers compartment only

This easiest fuselage configuration is the one in which the fuselage performs the previously identified primary function; thus, simply providing accommodation for passengers only. In this case, the pilots are not required and the unmanned vehicle is automatically piloted trough pre-loaded flight profiles or the vehicle is experiencing an un-controlled mission. Looking at this category, two possible applications of this elementary configuration can be envisaged. The first application would be a vehicle second stage aimed at performing a re-entry mission and the second application could be a second stage of a suborbital transportation system. This configuration is characterized by an extreme compactness that makes it suitable especially for limited number of passengers. Moreover, it can be easily designed and produced, with a consequent benefit for both logistics and maintenance activities. As far as safety is concerned, for this configuration there is not the need of envisaging a special escape system because the entire fuselage can be considered an escape system itself. The absence of a propulsion and propellant subsystems installed close to the passengers' compartment improves the level of safety by reducing the risk related to explosion. This fuselage configuration, as well as all the other ones presented in the following subsections, is in anyway preventing the presence of the any other subsystems within the vehicle, but they are not installed in fuselage. Figure 172 summarizes the major steps that should be performed in order to properly design a fuselage consisting of a passengers' compartment only.

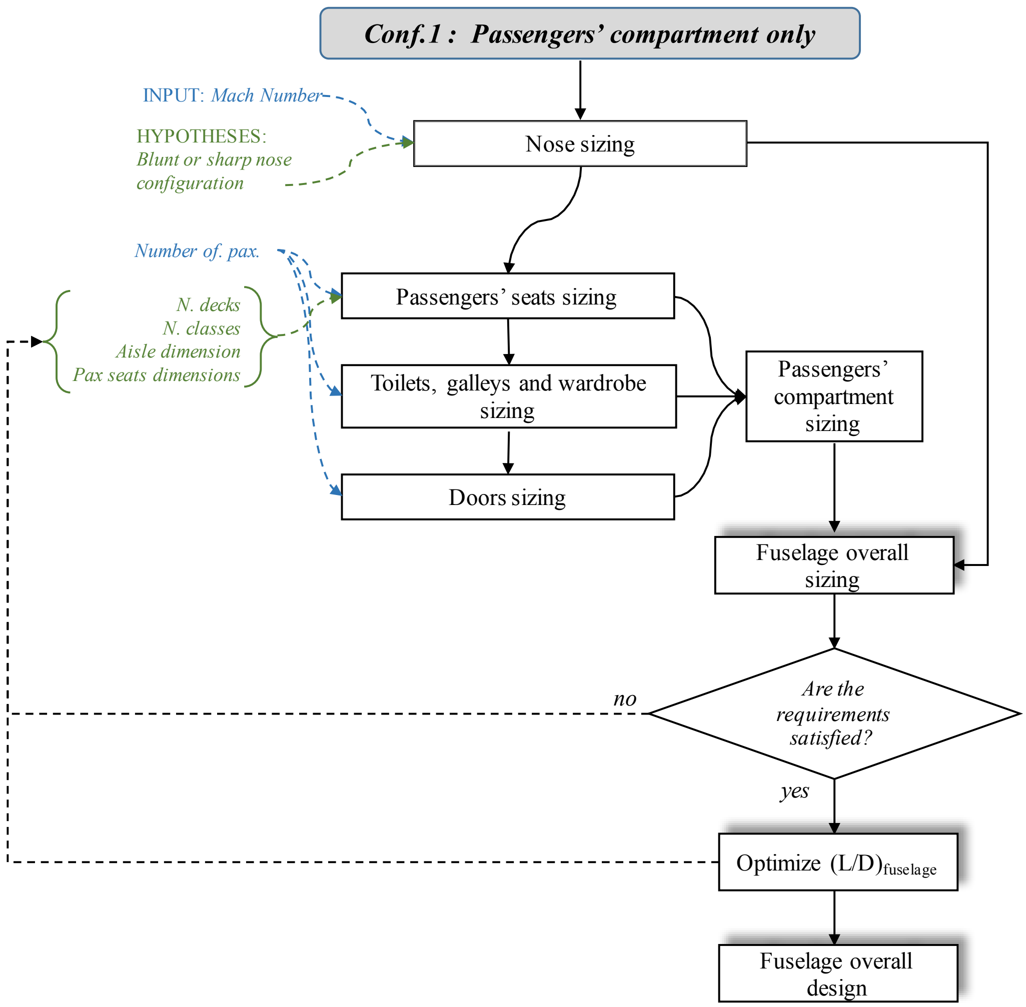


Figure 172: Flow-chart for the sizing of fuselage Conf.1.

6.3.2 Conf. 2: Crew and Passengers' compartment

This configuration is similar to the previous one with the additional presence of a crew compartment. This extension widely enlarges the number of enviable applications of this fuselage configuration. Indeed, it can be suitable for all those cases in which passengers should be transported and the propulsive and propellant subsystems are located in wing or anyway outside of the fuselage. The presence of a crew compartment ensures the possibility of controlling the vehicles and enlarge the possibility of applications beyond the un-guided re-entry. As it is explained in the previous section, the absence of a propulsion and propellant system installed closed to the passengers' compartment improves the level of safety by reducing the risk related to explosion. Figure 173, summarizing the major steps for the design of such fuselage configuration, differs from Figure 172 presenting two additional activities dealing with the design of cockpit and crew compartment.

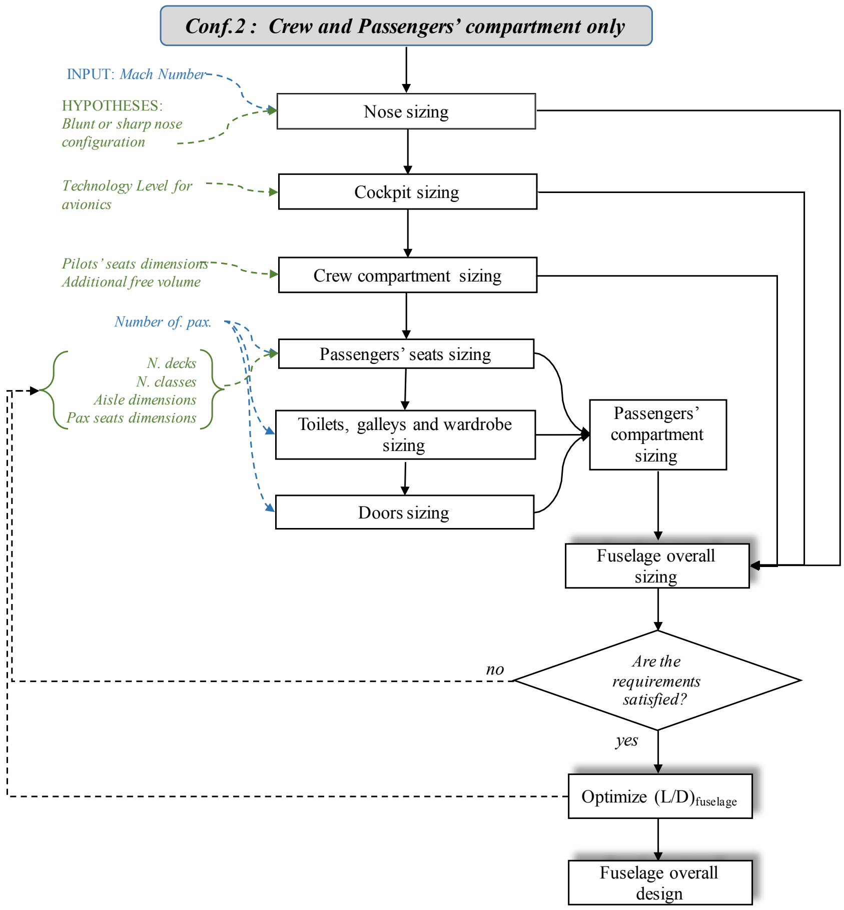


Figure 173: Flow-chart for the sizing of fuselage Conf. 2.

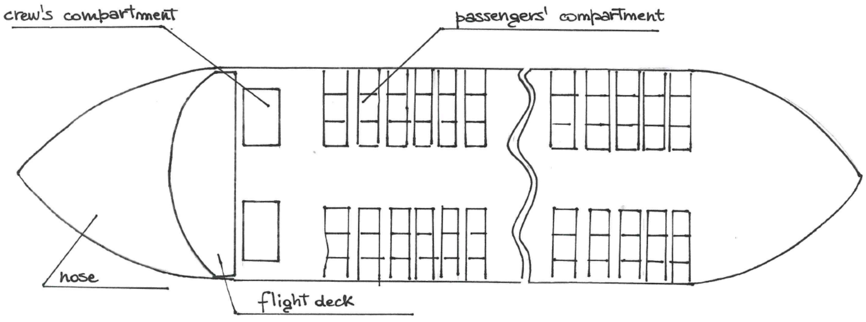


Figure 174: Qualitative sketch of a generic fuselage Conf. 2.

A qualitative sketch of a generic fuselage configuration witth both crew and passengers compartment is reported in Figure 174.

6.3.3 Conf. 3: Crew and passengers' compartments + air-breathing engines

This configuration is characterized by a higher level of complexity with respect to Conf.1 and Conf. 2, mainly due to the presence of a certain number of air-breathing propulsive units to be host in fuselage, with several implications on the design of the overall vehicle layout.

In order to properly locate the air-breathing engines within the under-development fuselage, the following major integration issues should be considered:

- Air-intakes design and location should be properly defined in order to maximize the performance of the air-breathing propulsion systems during the entire spectrum of operative speeds.
- Exhaust ducts should be located and arranged in such a way that they can guarantee vertical take-off and landing strategies, when required.

As far as the engine inlets are concerned, different solutions may be envisaged (Figure 175)

- Single integrated inlet (upper fuselage)
- Single integrated inlet (lower fuselage)
- Split integrated inlet placed on the lateral surface of the fuselage.

Each of these alternatives has different advantages and disadvantages that are all relative to the specific mission requirements that will lead each single design activity. Complementary, the shape of the inlet will depend on the performances required to the propulsive system and in particular to maximum operative speed that will be envisaged. Furthermore, as far as the exhaust duct is concerned, it is important to notice that additional difficulties may arise in case VTOL capabilities would be required. In this case, depending on the type of selected propulsive strategy to perform the vertical take-off and landing, both the location and sizing of the engine exhaust ducts and nozzles can have a deep impact on overall layout configuration.

Figure 176 summarizes the major steps to be carried out in the design of a fuselage integrating an air-breathing propulsion system.

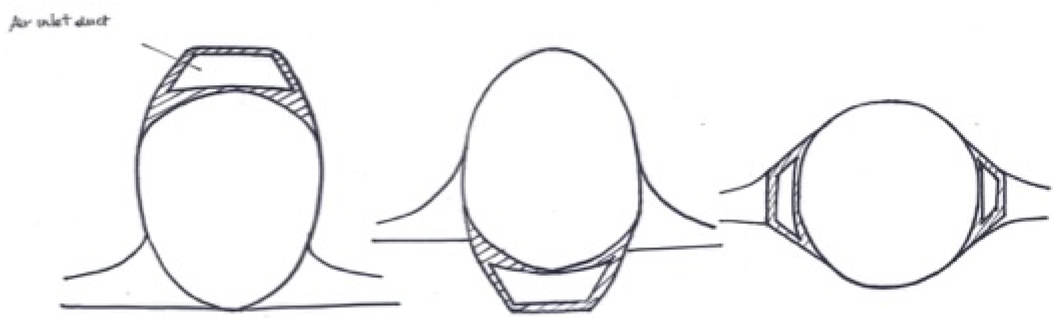


Figure 175: Possible air inlet location

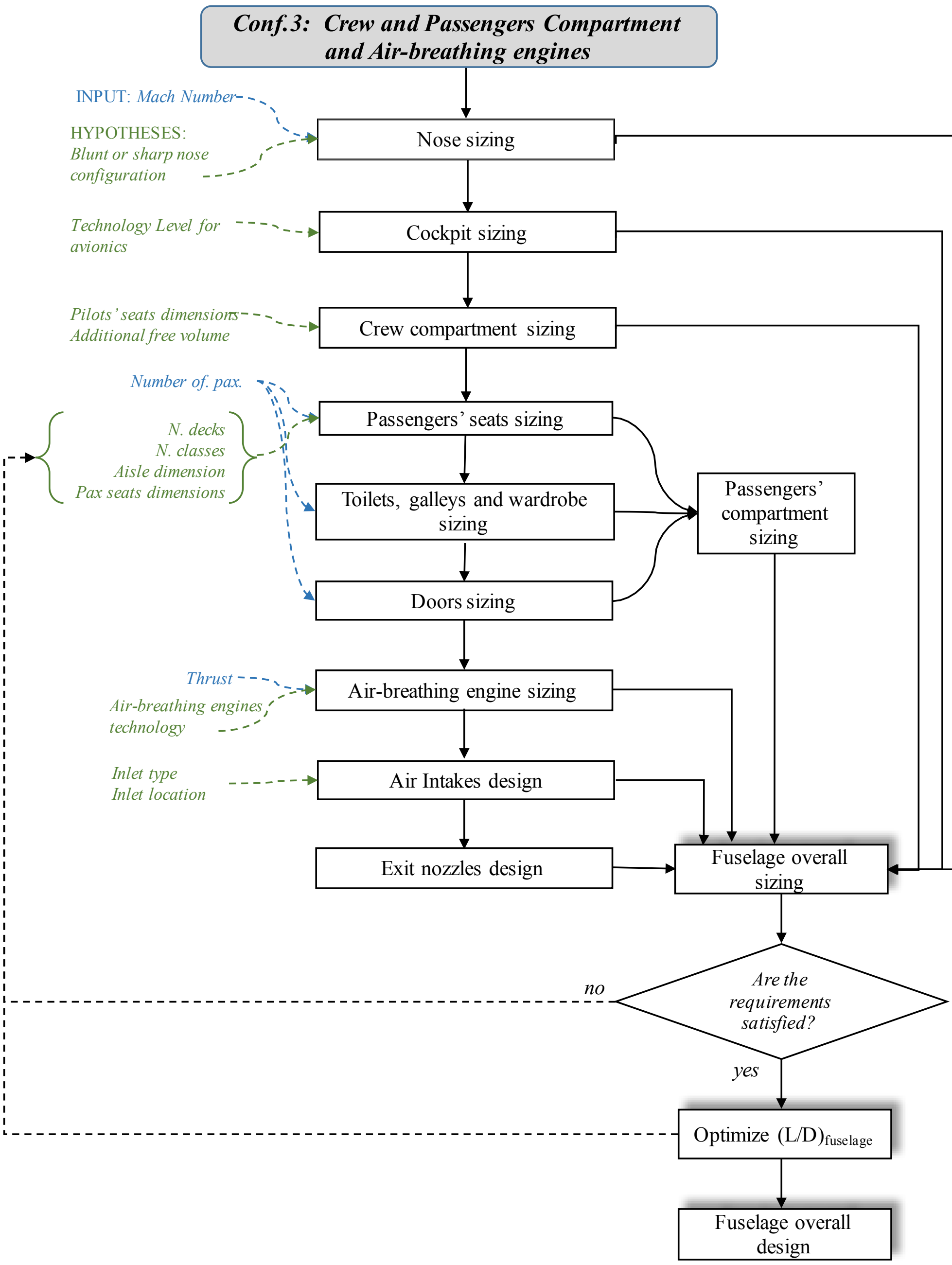


Figure 176: Flow-chart for the sizing of fuselage Conf. 3.

6.3.4 Conf. 4: Crew and passengers' compartments + rocket motor

In case of rocket motor, the integration within the fuselage is easier than in the previous case, mainly because of the absence of an inlet, deeply impacting on the configuration. The major problems related to this type of configurations are linked to safety considerations that would prevent the designer to place the propulsive motor, too close to the crew and passengers' compartment.

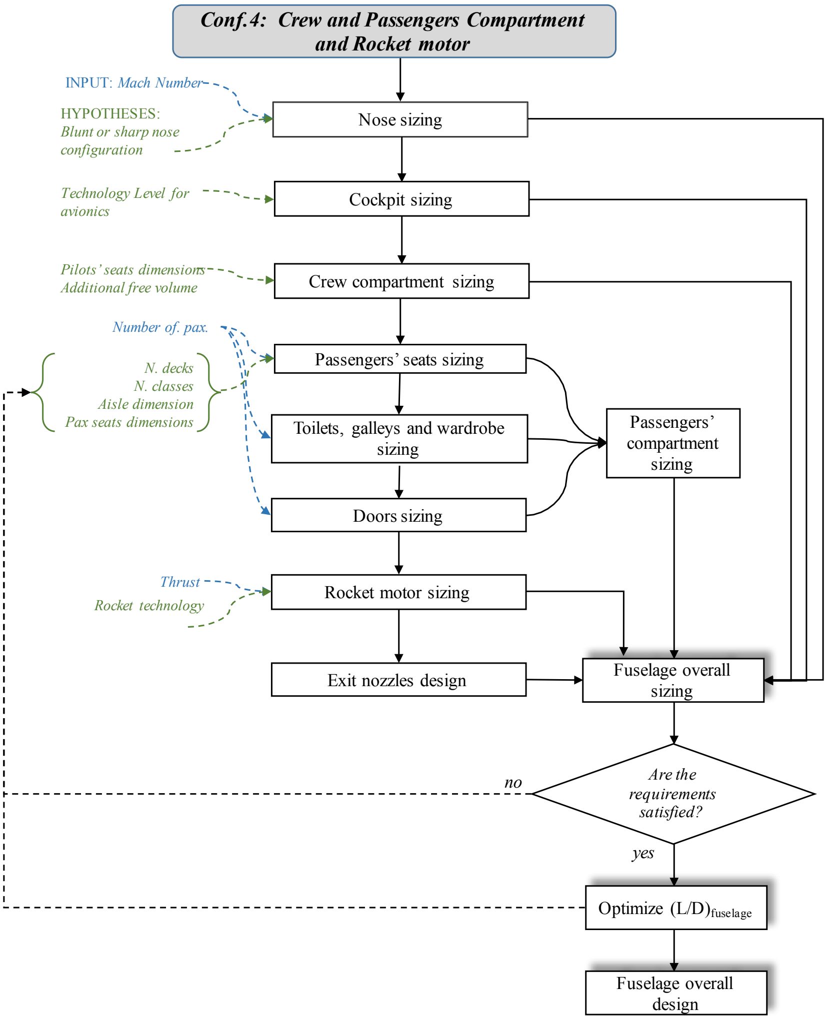


Figure 177: Flow-chart for the sizing of fuselage Conf. 4

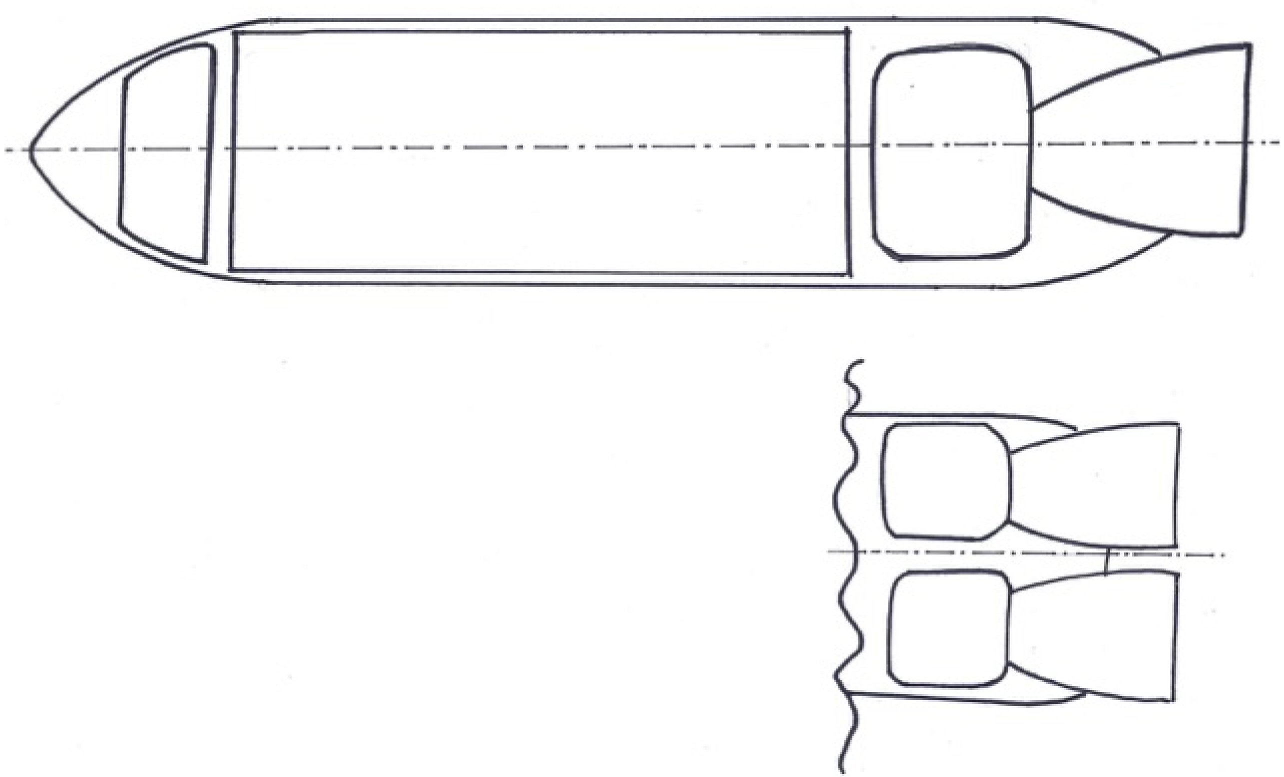


Figure 178: Qualitative sketch of a generic fuselage Conf. 4.

6.3.5 Conf. 5: Crew and passengers' compartments + rocket+ propellant tanks

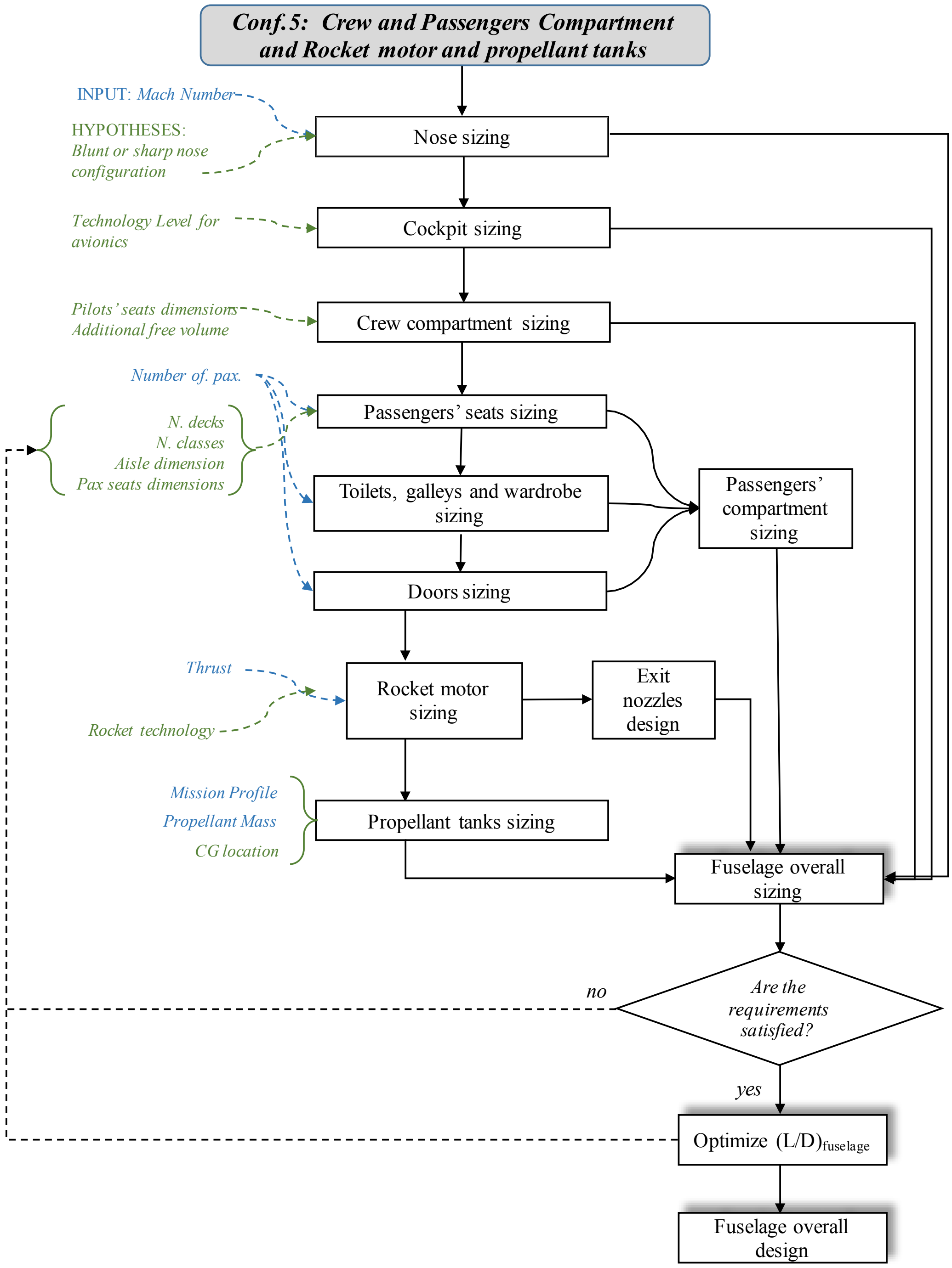


Figure 179: Flow-chart for the sizing of fuselage Conf. 5

Together with Conf. 7, this is one of the most complex and dangerous configuration mainly due to the presence of propellant tanks on-board. On the other hand, it is the most advanced and probably compact configuration and thus, very slender layout may be adopted.

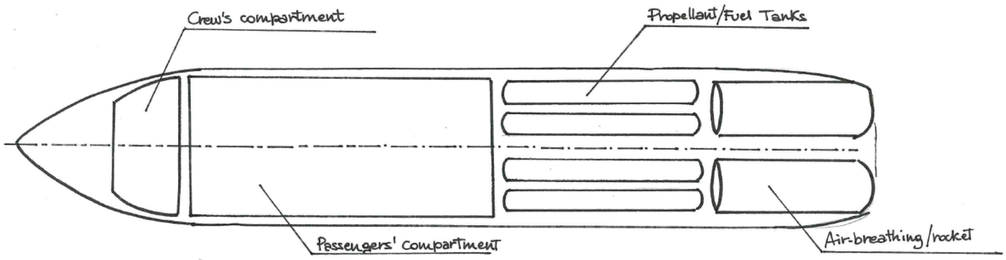


Figure 180: Qualitative sketch of a generic fuselage Conf. 5.

6.3.6 Conf. 6: Crew and passengers' compartments + air-breathing engines+ fuel tanks

This configuration describes a special condition in which, the room available in wing is no more sufficient and there is the need of hosing fuel tanks in fuselage. This is not an uncommon choice in military aircraft even if this is strongly discouraged by civil transportation regulation. However, this regulation could be too much restrictive for the case of hypersonic transportation systems. Considering the possible location of these additional tanks, proper room, under the passengers' compartment, closed enough to the aircraft CG, in order to minimize the CG excursion due to the fuel depletion during the mission, can be envisaged. In this case, the location of main landing gear when retracted, whether located in fuselage should be properly assessed. Another criterion that could be considered to properly located these tanks is to shorten as much as possible the feeding lines, thus to select a location closer to the engine combustion chamber.

It is clear that all the suggestions related to engine inlet and outlet location and sizing proposed for Conf. 3 are still valid and can be exploited for this configuration too.

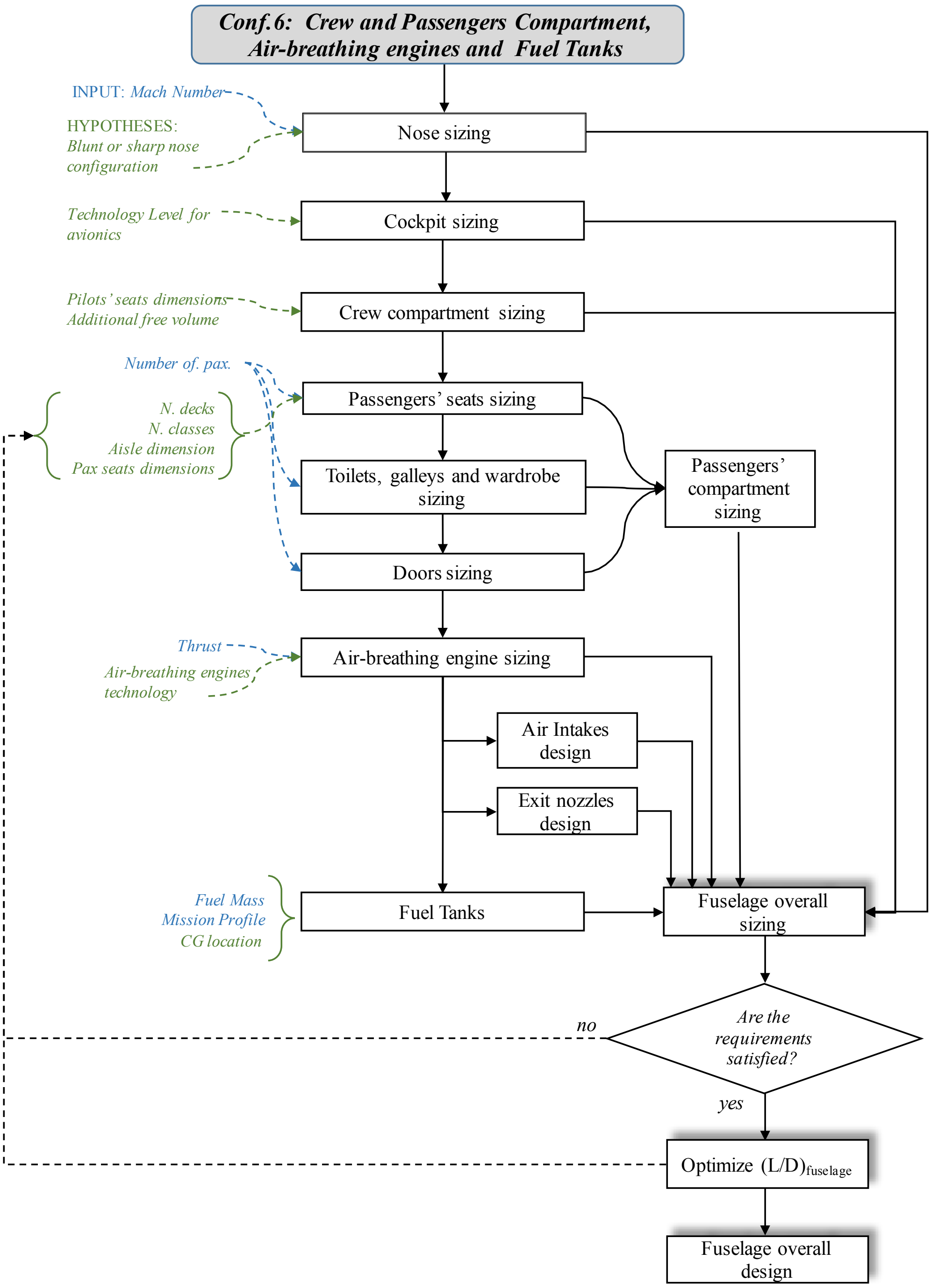


Figure 181: Flow-chart for the sizing of fuselage Conf. 6

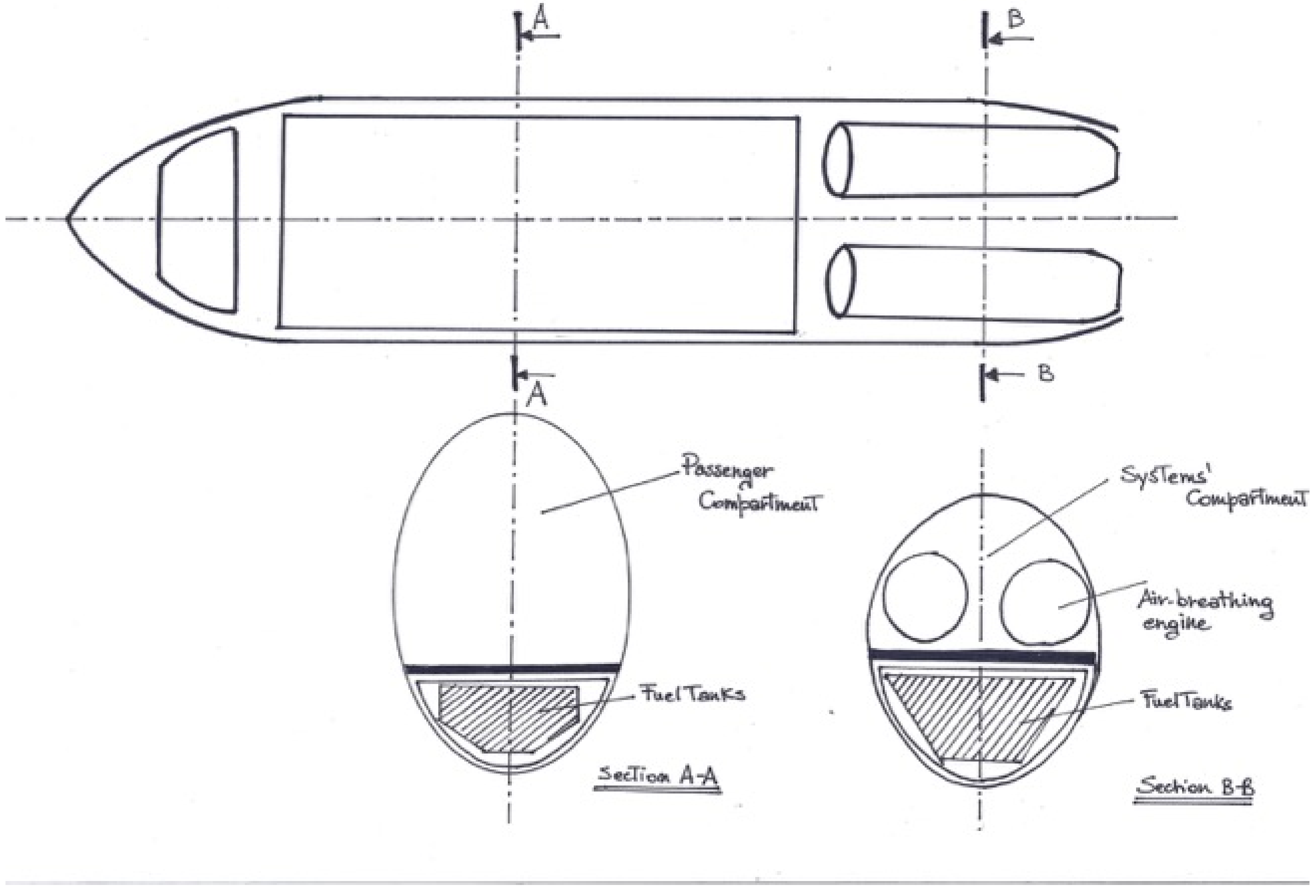


Figure 182: Qualitative sketch of a generic fuselage Conf. 6

6.3.7 Conf. 7: Crew and passengers' compartments + air-breathing engines +rocket motor

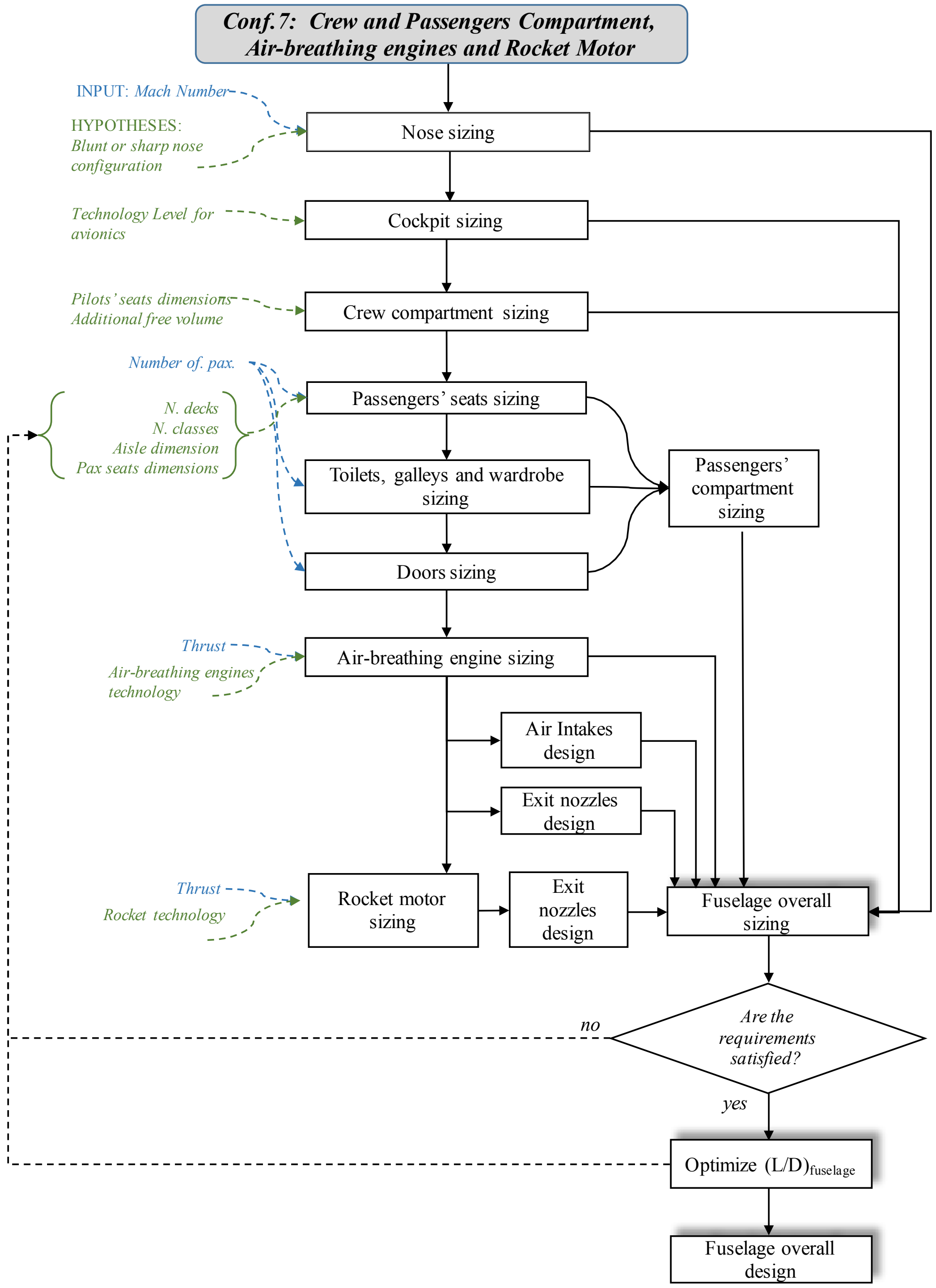


Figure 183: Flow-chart for the sizing of fuselage Conf. 7

This configuration envisages the mutual presence of both propulsion subsystems while the related fuel and propellant tanks are hosted far away, probably hosted in the available room in wing. Considering the need of ensuring a cabin escape system, and the problem of integrating the exhaust hot gases ducts, most likely, both the air-breathing and the rocket engines could be located in the aft fuselage, in a tandem configuration. However, the configuration is also depending on the number of propulsive units requires. For example, in case of a high number of engines, a propulsive fuselage strategy may be adopted. Please, notice that thrust symmetrical conditions may be guaranteed. In this case, as well as in all the other configurations envisaging the presence of air-breathing engines, proper air-intakes must be exploited.

It is clear that all the suggestions related to engine inlet and outlet location and sizing proposed for Conf. 3 are still valid and can be exploited for this configuration too.

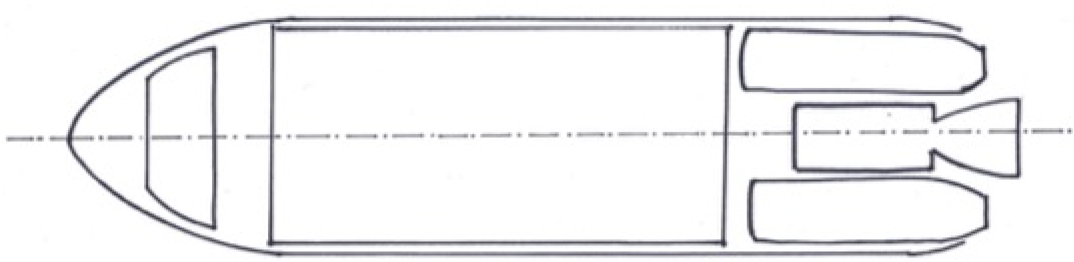


Figure 184: Qualitative sketch of a generic fuselage Conf. 7

6.3.8 Conf. 8: Crew and passengers' compartments + air-breathing engines +rocket motor + fuel tanks and propellant tanks

This is the most complex enviable configuration because it has both the propulsion systems types and the related tanks. In this case, the most likely configuration will be to place the propulsive units at the end of the aft fuselage section, placing the propellant tanks between the passengers' compartment and the propulsive units. Complementary, the fuel tanks can be located under the floor of the passengers' compartment in a location as much as possible closer to the overall aircraft configuration CG. It is clear that all the suggestions related to engine inlet and outlet location and sizing proposed for Conf. 3 are still valid and can be exploited for this configuration too.

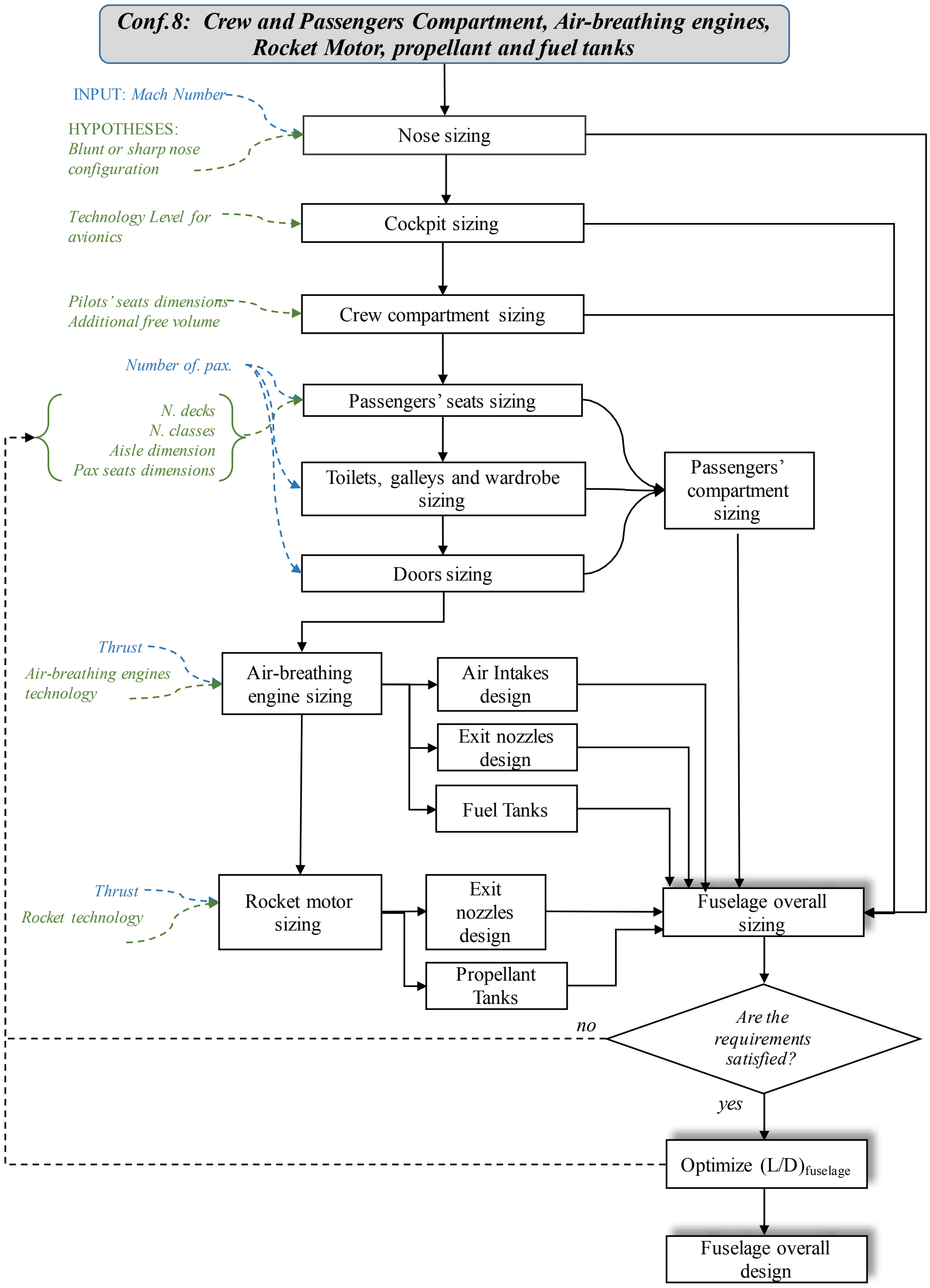


Figure 185: Flow-chart for the sizing of fuselage Conf. 9

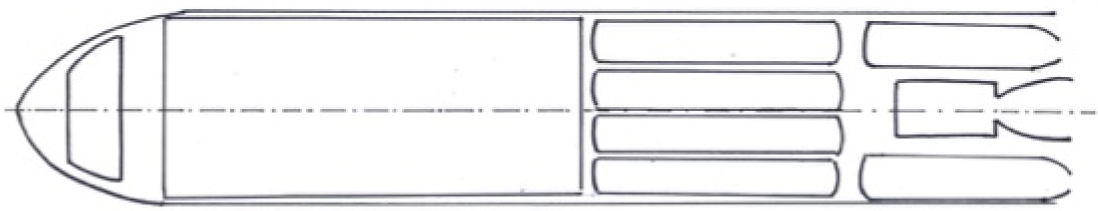


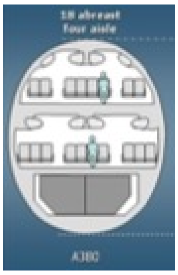
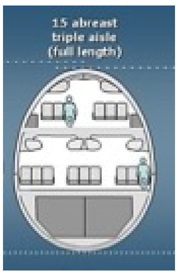
Figure 186: Qualitative sketch of a generic fuselage Conf. 8

6.3.9 Fuselage Section configuration

Considering the shape of the Fuselage section, especially in the part of the Passengers' compartment, and mainly focusing on hypersonic vehicles, the circular or elliptical configuration seems to be the only feasible one, for several reasons, including aerothermodynamic issues and structural ones. The internal layout, especially of the passengers' compartment can deeply affect the external shape of the fuselage. In particular, depending on the number of passengers, the designer shall understand whether a double deck configuration will be useful. Indeed, in case of double deck configuration, the fuselage section shall be quite far from a pure circular or elliptical one but other shapes are preferred such as ovoid and bi-lobe.

In the following table, the major advantages and disadvantages of some enviable fuselage section shapes with the indication of existing vehicles whose vehicle passengers compartment has this internal layout.

Table 88: Configuration alternatives for fuselage section.

	Passenger compartment cross section shape	***Characteristics***
Circular		• Reduced structural weight • Room available for both passengers and cargo • Simple but effective shape • Fair exploitation of the available room
Bi-lobe		• Optimized class subdivision. • Room available for both passengers and cargo • Complex shape that may have negative impacts on aerodynamics and structure. • Complexity in production and integration • Good exploitation of the available room
Elliptical		• Double-deck possibility • Good exploitation of the available room • Optimal configuration for narrow body configuration • Simple shape with positive impact on aerodynamic, structure, production and integration
Ovoid		• Double-deck possibility • Optimal exploitation of the available room • Optimal configuration for narrow body configuration • Simple shape with positive impact on aerodynamic, structure, production and integration

6.4 Fuselage sizing

Once the high-level fuselage configuration has been sketched, both in terms of section layout and on-board systems hosted inside the structure. In order to carry out the estimation of the overall fuselage length, the activity flow reported in the several flow-charts specifically derived for each fuselage configuration can be applied. The overall fuselage length estimation can be carried out quite early during the design process and with a good confidence level. This is possible first of all thanks to a high level of standardization of the several elements involved, mainly due to existing regulations (and so, strictly related to safety concerns) or guidelines to ensure proper comfort level. This is evident in the sizing algorithm suggested for the crew and passengers' compartments. It has to be notice that, besides the fact that there is not a specific regulatory framework for hypersonic and suborbital vehicle, the Certification Specification CS 25 (EASA, 2017) has been considered a valuable reference.

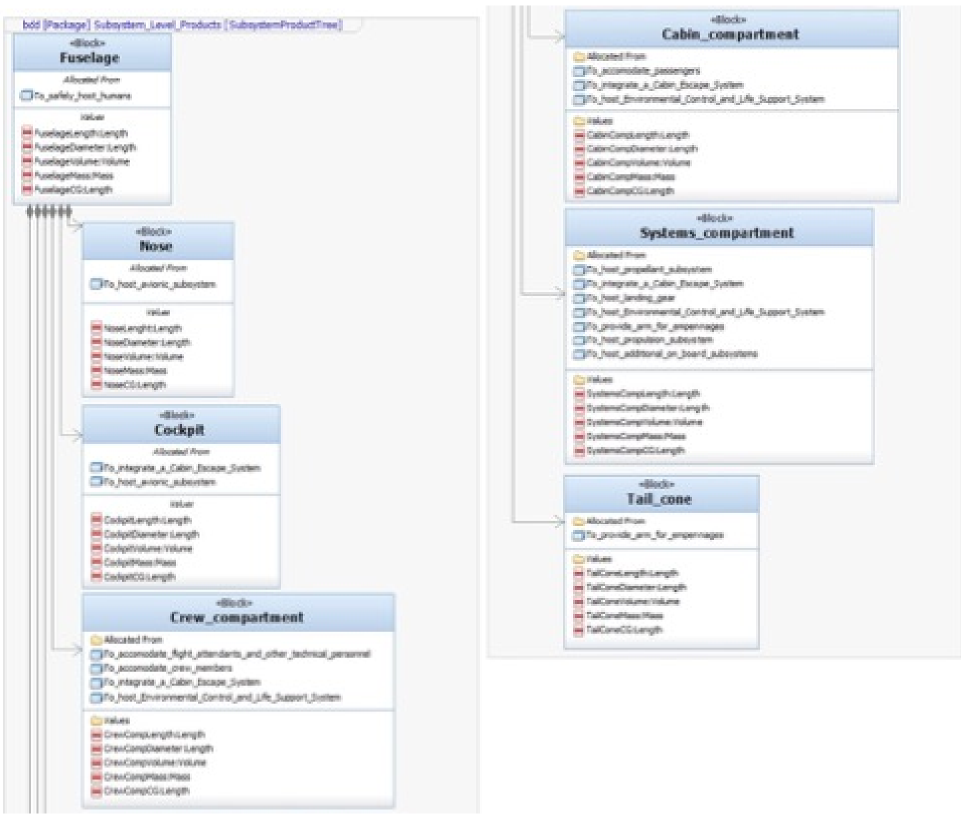


Figure 187: Fuselage BDD with functions allocation and products associated descriptive parameters

6.4.1 Front Fuselage sizing: nose, cockpit and crew compartment

With the term *front fuselage* is here intended to consists of:

- Nose section
- Cockpit section
- Crew compartment section.

as it is qualitatively sketched in Figure 188.

The overall sizing algorithm suggested in the following subsections, will allow to define the overall front fuselage layout with a first sizing attempt.

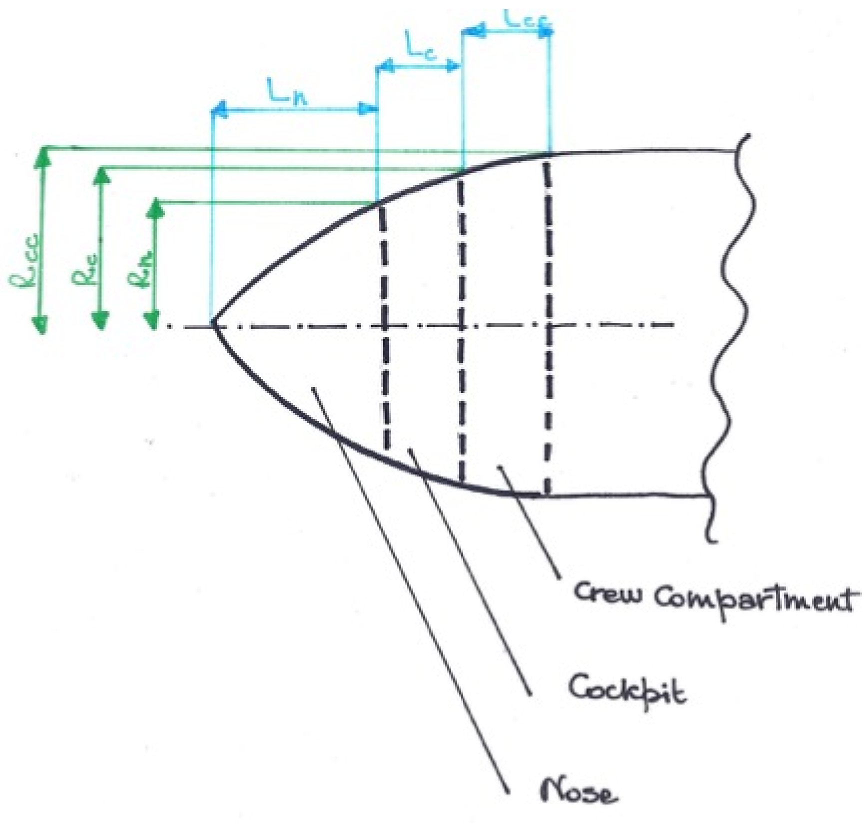


Figure 188: Generic Sketch of the forward fuselage section components

6.4.1.2 Fuselage Nose Sizing

Fuselage Nose Section Sizing Algorithm

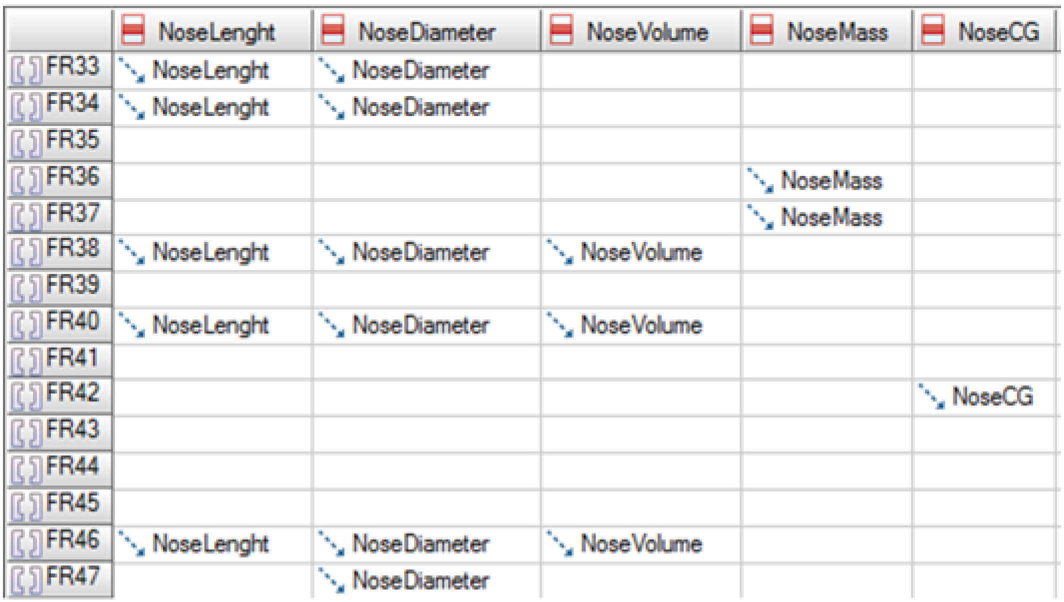
The fuselage nose design is mainly affected by the need of guaranteeing proper aerodynamic and aerothermodynamic characteristics and the logistic and operational related needs, mainly related to guarantee pilots proper visibility. These considerations are perfectly stored in the influence matrix reported in Table 89. Indeed, this matrix can be very useful to understand the impact of each requirement to the fuselage nose sizing and more important to the most important design variables (Table 90 and Figure 189).

Table 89: Requirements impact on fuselage nose section.

		Requirements impacting on fuselege nose section design	*Impacting Drivers*	*Impacted design parameters*	*Comments*
Comfort and Safety	*The fuselage shall safely accommodate crew members*				
	The fuselage shall safely accommodate passengers				
	The fuselage shall guarantee proper room allowing passengers to experience microgravity				
	The fuselage shall guarantee proper room allowing scientists to carry out experiments.				
	The fuselage shall guarantee a proper view of the Earth.				
Aerodynamic	*The fuselage shall generate the lowest possible drag.*		• Mach Number • Layout configuration	L_nose = f(μ; k_nose); D_nose = f(μ)	The drag produced by the fuselage nose section can be minimized sketching the nose section in such a way that it is completely wrapped by the Mach cone, whose aperture is defined by μ = asin(1/Mach). In addition, a proper shape of the front nose part can be selected, moving towards sharper
	The fuselage shall contribute positively to the lift generation				
	The fuselage heat loads shall be minimized		• Mach Number • Layout configuration	L_nose = f(μ; k_nose); D_nose = f(μ)	The heat loads experienced by the fuselage nose section can be minimized sketching the nose section in such a way that it is completely wrapped by the Mach cone, whose aperture is defined by μ = asin(1/Mach). Unfortunaly, this is not sufficient and a proper radius for the front nose part shall be selected
	The fuselage wetted area shall be minimized.				
Structure	*The fuselage weight shall be minimized.*		• Avionic equipment • Material • Nose wet surface • TPS thickness	m_nose = f(S_nose; t_wall; m_avionic)	The nose fuselage mass shall be properly allocated to the
	The fuselage shall sustain the structural loads all along the flight profile.				
	The fuselage shape shall be as symmetric as possible.		Mach Number	Nose section shape	The nose section shal be as symmmetric as possible, meaning that the center of the section shall be as close as possible to vehicle longitudinal axis. This will not numerically impact on nose length or diameter directly, but it has a primary influence on the
	The fuselage shall accommodate landing gear.				
	The fuselage shall accommodate propulsion systems.				
	The fuselage shall accommodate avionic subsystems		Presence of avionic apparati to be installed in the fuselage nose section.	V_nose = f(V_avionic); L_nose = f(k_nose_add); D_nose = f(D_avionic)	The nose volume (V_nose) shall be sufficiently wide to host the avionic equipmens that should be installed inside. In this context, also additional paykoad such as monitoring devices or radars shall be taken into account.
	The fuselage shall accommodate propellant subsystem				
	The internal arrangement shall guarantee the proper centre of gravity location.		Integration of avionic equipment	CG_nose = f(m_avionic)	The center of graviti position of the nose section is mainly dependent from the integration of the avionic equipment installed
	The overall fuselage structure shall be able to separate the cabin escape system.				
	The fuselage shall properly accommodate cargo				
Logistics and Operations	*The fuselage shall ease loading and unloading operations*				
	The fuselage shall guarantee proper airworthiness characteristics.		Regulations	Nose section shape (k_nose)	The shape of the nose fuselage section shall be properly hypothesized considering exiting airworthiness regulations
	The fuselage shall be properly designed in order to allow visibility to the pilots.		• Mission profile	α_nose = f(α_approach; V_approach)	The overnose vision shall be guaranteed. This design parameter can be evaluated on the bases of mission trajectory data such as the ange of attack (α_approach) and the speed during approch phase (V_approach).
	The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.		• Mission profile • Emergency scenarios	Nose section shape and integration	In case of emergency, the possibility of dtaching part of the fuselage shall be envisaged, in order to diminish the risk of loss

Table 90: Requirements allocation on the main fuselage nose section design parameters.

	Nose Length	Nose Diameter	Nose Volume	Nose Mass	Nose CG
The fuselage shall safely accommodate crew members					
The fuselage shall safely accommodate passengers					
The fuselage shall guarantee proper room allowing passengers to experience microgravity					
The fuselage shall guarantee proper room allowing scientists to carry out experiments.					
The fuselage shall guarantee a proper view of the Earth.					
The fuselage shall generate the lowest possible drag.					
The fuselage heat loads shall be minimized					
The fuselage wetted area shall be minimized.					
The fuselage weight shall be minimized.					
The fuselage shall sustain the structural loads all along the flight profile.					
The fuselage shape shall be as symmetric as possible.					
The fuselage shall accommodate propulsion systems.					
The fuselage shall accommodate avionic subsystems					
The fuselage shall accommodate propellant subsystem					
The internal arrangement shall guarantee the proper centre of gravity location.					
The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.					
The fuselage shall properly accommodate cargo					
The fuselage shall ease loading and unloading operations					
The fuselage shall guarantee proper airworthiness					
The fuselage shall be properly designed in order to allow visibility to the pilots.					



	NoseLenght	NoseDiameter	NoseVolume	NoseMass	NoseCG
FR33	NoseLenght	NoseDiameter			
FR34	NoseLenght	NoseDiameter			
FR35					
FR36				NoseMass	
FR37				NoseMass	
FR38	NoseLenght	NoseDiameter	NoseVolume		
FR39					
FR40	NoseLenght	NoseDiameter	NoseVolume		
FR41					
FR42					NoseCG
FR43					
FR44					
FR45					
FR46	NoseLenght	NoseDiameter	NoseVolume		
FR47		NoseDiameter			

Figure 189: MBSE implementation of requirements allocation on the main fuselage nose section design parameters.

Before entering in the detail of the evaluations, Figure 190 reports the activity flow for the sizing of the fuselage section.

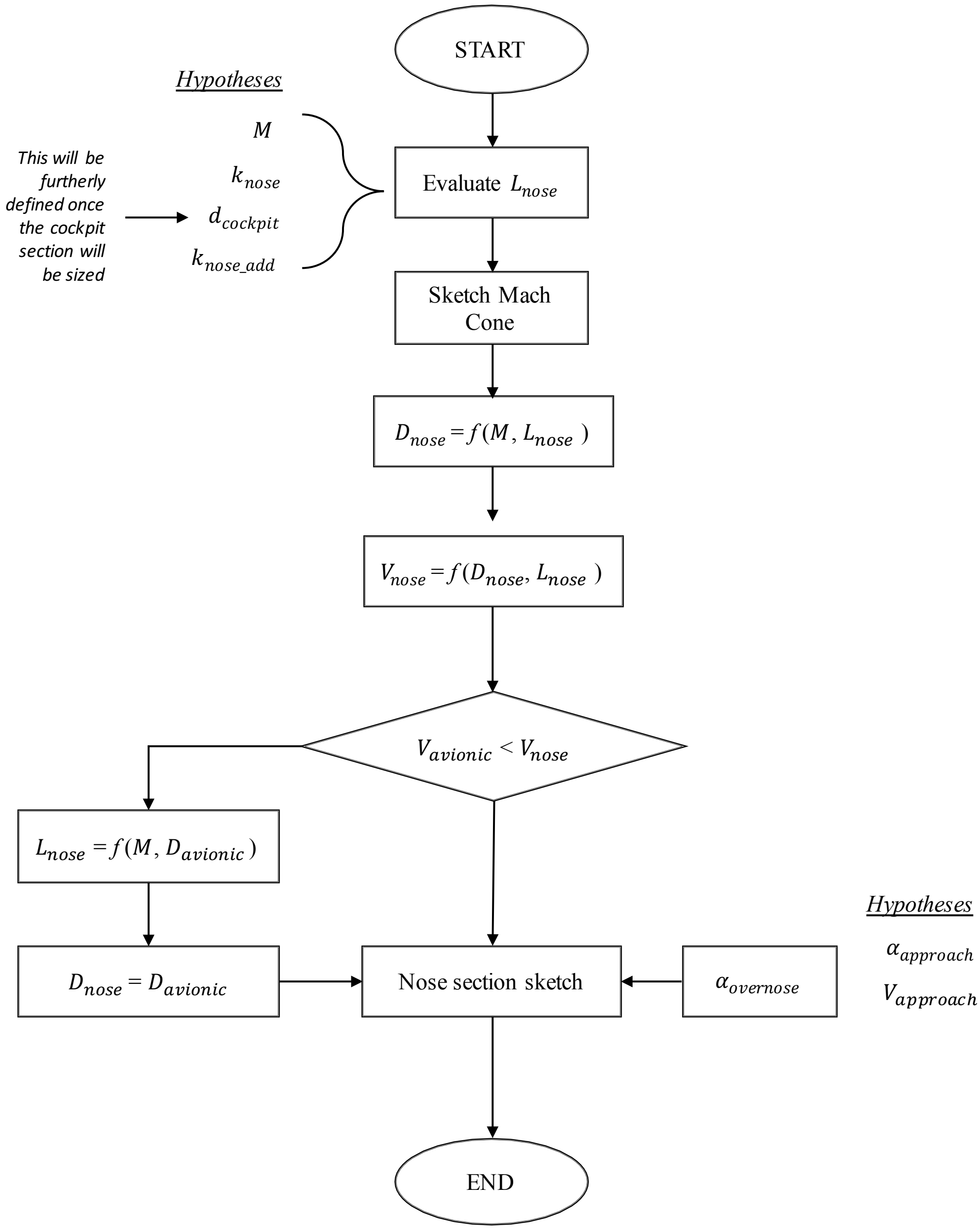


Figure 190: Flow-chart summarizing the sizing algorithm suggested for the fuselage nose section.

Mathematical details for the fuselage nose section sizing

The nose of the vehicle is one of the part of the vehicle whose shape and layout is strongly affected by aerodynamic and aerothermodynamic considerations. This is the reason why, in order to properly define the nose layout, the Mach cone shall be considered as a guideline. Thus, as a first estimation attempt, the length of the vehicle nose can be estimated on the basis of trigonometric rules, with the introduction of a corrective coefficients assuming different values depending on the fact that different nose layout can be selected, from sharp to blunt profiles.

$$L_{nose} = \frac{D_{cockpit}}{2\ \tan\left(\frac{\mu}{2}\right)} \cdot \frac{1}{(1 + k_{nose})} + k_{nose\ add}$$

where

k_{nose} is a corrective factor that can range from 0 (sharp configuration) and 1 (blunt configuration) and express the percentage of the ideal length of the Mach cone.

k_{nose_add} is a design margin considered in order to take into account the possible need of exploiting the nose part of the fuselage to host avionics

$d_{cockpit}$ is the cockpit diameter.

$\mu = \sin^{-1}\left(\frac{1}{Mach}\right)$ is the Mach cone semi-aperture.

This first estimation is based on pure aerodynamic considerations. Actually, the estimation shall take into account the need of integrating additional subsystems in this front part. A clear example could be the integration inside the nose of a radar and of avionic apparatus such as the EO/IR turret aimed at performing surveillance or monitoring activities. In this case, proper integration strategies should be considered (Fusaro, 2015). In this context, the integration of innovative sensors based on hyperspectral technology can be considered also for hypersonic and suborbital flights considering that this technology is currently exploited in both aeronautical and space applications. Moreover, the design of the vehicle nose may also undergo additional considerations mainly related to aerothermodynamic characteristics and pilot visibility.

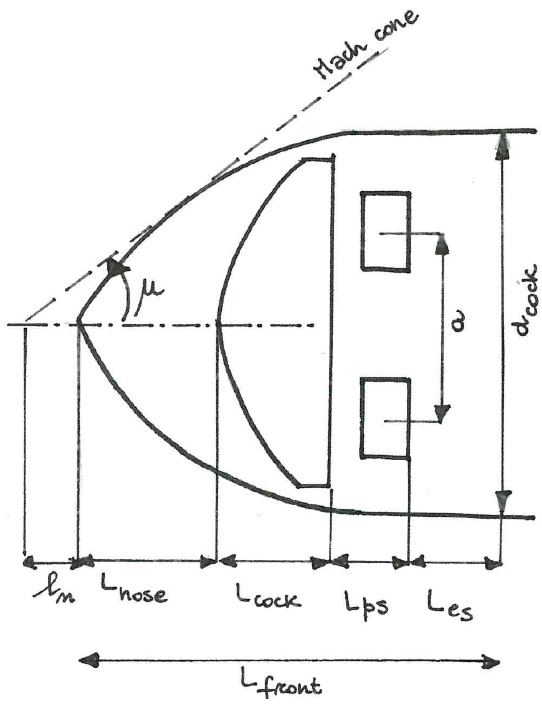


Figure 191: Guidelines for the forward fuselage sizing

Complementary, the Mach number has also impact on the side view definition even if, visibility requirements have the major impact on this sizing activity. Indeed, a proper pilot view shall be guaranteed all along the mission but especially during take-off and landing. In this context, it is also convenient to consider that this procedure may also be used for VTOL aircraft; indeed, following these design guidelines allow envisaging a vehicle that in case of contingency, could hypothetically perform traditional take-off and landing operations.

Figure 192 shows that the semi-aperture of the Mach cone is not the only sizing requirement to be considered but that it shall be compared with the angle ensuring a good visibility to the pilots. Indeed, the overnose vision is critical for safety especially during landing. Considering available data, general aviation aircraft land in a fairly level attitude and so have small overnose vision angles (5-10 deg), while civilian transports have in general much wider overnose vision angles (up to 25 deg).

In particular, the angle resulting from the pilots' eyes line and the aircraft nose can be estimated as follows (Raymer, 2012).

$$\alpha_{overnose} = \alpha_{approach} + 0{,}04 \cdot V_{approach}$$

where

$\alpha_{approach}$ is the angle of attack of the aircraft during approach procedure;

$V_{approach}$ is the aircraft approach speed.

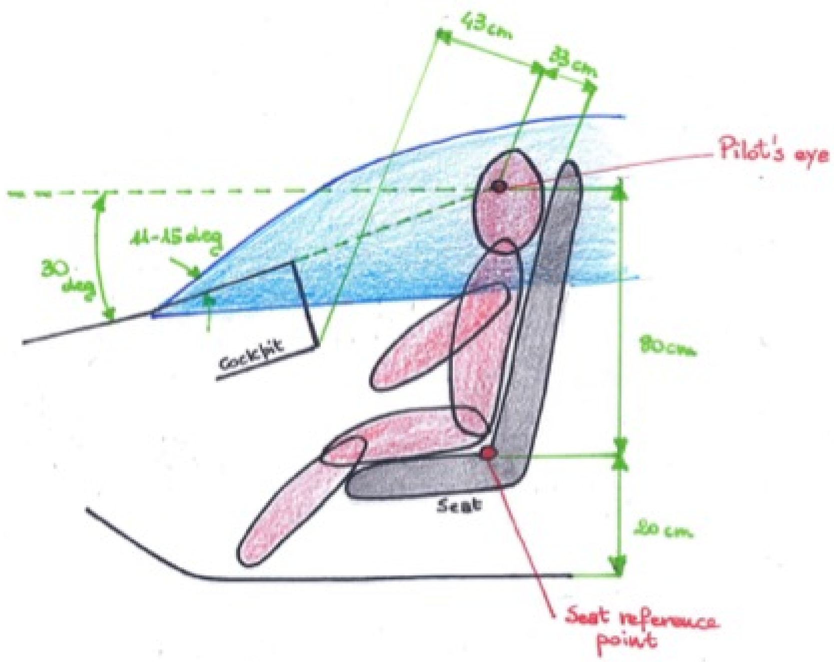


Figure 192: Guidelines for the side view definition of the forward fuselage section.

Entering in the detail of the section view of the nose, in a first estimation, it can be defined geometrically, hypothesizing a conical shape whose section diameter can be defined by the Mach cone aperture and but even more, by the need of on-board installing some avionic equipment as well as the need of introducing payloads, like sensors, able to perform monitoring activities. This is the special case of hyperspectral sensor for example, a sensor based on innovative technologies that is currently exploited on both aeronautical and space platforms. (Fusaro, 2015) (Fusaro, 2016)

Looking at the scheme reported in Figure 191, the following equations can be applied in order to define the section of the fuselage nose. For the purposes of the conceptual design activities, the fuselage nose can be simplified by means of a conical shape. Besides not being so realistic, the dimensions obtained with this approach are not so far from the real ones. However, it has to be noticed that in any case, the nose conical shape must suit within Mach cone. For this reason, an iterative procedure is here suggested allowing an optimal fulfilment of the requirements. In particular, as it is shown in the following scheme, knowing the nose length, it is possible to evaluate the diameter of the ideal cone at the end of the fuselage nose section. Then, it has to be considered if the volume required by the avionic bay or the sensor can be accommodated within the Mach Cone, taking into account proper design margins. In case of feasible integration, the Mach Cone can be considered as the external envelope for the fuselage nose section. Conversely, in case this additional volume will be required, the only possible solution is to stretch the configuration, envisaging a longer fuselage nose section. The new estimation of nose length can be carried out hypothesizing the diameter of the ending section equal to the diameter required to host avionics inside and then, to re-evaluate the length of the Mach cone required to have this ending diameter. The following equations can be applied to perform the estimations. In particular, the second one has been obtained properly modifying a generic cone equation, to the purposes of this section, where the maximum cone diameter can be defined as:

$$D_{nose} = 2 \cdot \mathrm{L_{nose}} \tan\left(\frac{\mu}{2}\right)$$

$$\frac{\mathrm{z}^2 + \mathrm{y}^2}{\left(\frac{\mathrm{D_{nose}}}{2\mathrm{L_{nose}}}\right)^2} = \mathrm{x}^2$$

where

$\mu = \sin^{-1}\left(\frac{1}{Mach}\right)$ is the Mach cone semi-aperture.

x, y and z are the coordinates used to generate the cone. The reference system, as highlighted in Figure # has its origin in the edge of the Mach Cone.

Then, in order to increase the level of confidence of the performed estimation, some corrections can be introduced, taking into account constraints mainly related to aerothermodynamic. A proper radius shall be selected and its reference point located. From these hypotheses it is possible to draw a more accurate sketch of the fuselage nose. Eventually, on the basis of all these preliminary evaluations, it is possible to derive a first estimation of the wetted surface of this fuselage part, simply on the basis of geometrical considerations.

Weight and Balance considerations for the fuselage nose section

Once the fuselage nose section has been defined in terms of layout and a preliminary sizing has been completed, a first mass and weight & balance estimation can be carried out. This activity is fundamental for a correct integration of the fuselage into the vehicle. To do this, it is essential to have an idea of the different materials that may be implied in the nose. In particular, thinking that this part of the vehicle shall be properly protected by the heat loads, the TPS technology selection can have a deep impact on the mass of the fuselage nose. By the way, this is not the only contribute in terms of mass. It has also to be considered that additional mass shall be accounted for the presence of avionic equipment or sensors and related bay. In this context, it is useful to notice that in case of optical instruments, requiring structural apertures and mechanisms to close the holes during the most critical mission phases, an increment in the structural mass of the fuselage nose shall be considered.

$$m_{nose} = \sum_{i=1}^{n_s} \left(S_{wet_i} \cdot t_i \cdot \rho_i \right) + \sum_{j=1}^{n_e} m_{avionic\ bay_j} + k_{si}$$

where n_s is the number of sections with different materials

S_{wet} is the wet surface of each ith-esim section;

t is the wall thickness of each ith-esim section;

ρ_i is the density of each ith-esim section;

n_e is the number of avionic equipment to be installed in the fuselage nose section;

k_{si} is the coefficient taking into account additional mass due to systems integration.

Table 91: Useful suggestions for geometrical characteristics of typical shapes implied for the conceptual design definition of a fuselage

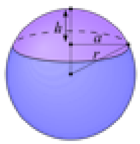
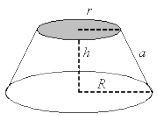
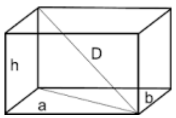
Section sketch	***Wet Surface***	***Volume***	***CG_evaluation***
Sphere 	$S = 2\pi R_1 h$ $h = R_1 - \left(\frac{r_1}{\tan\left(\frac{\theta}{2}\right)}\right)$	$V = \pi h^2 \left(r - \frac{h}{3}\right)$	$x_{CG} = \frac{1}{2}(R_1 + h)$
Cone 	$S = \pi a(R_1 + R_2)$	$V = \frac{1}{3}\pi h(R_1^2 + R_2 R_1 + R_2^2)$	$x_{CG} = \frac{h}{4}\frac{(R_1^2 + 2R_1R_2 + 3R_2^2)}{(R_1^2 + R_1R_2 + R_2^2)}$
Boxes 	$S_{lat} = 2(a+b)h$	$V = abh$	$x_{CG} = \frac{h}{2}$

Table 92: Useful data for the hot structure materials

Material	*Temperature*	*Density*
UHTC (Ultra High Temperature Ceramics) ZrB_2 **(Zircionium diboride)**	T >2000°C	6 g/cm^3
UHTC (Ultra High Temperature Ceramics) $HfBr_2$ **(Hafnium diboride)**	T >2000°C	10 g/cm^3
CMC (Ceramic Matrix Composites)	T>1300°C	15 kg/m^2

Table 91 collects practical formulas to evaluate the wet surface. In addition, a table with some technologies for thermal protection are reported with the aim of giving the readers practical suggestions to carry out these evaluations (Table 92).

Once the major mass properties of the section are known, the CG position can be evaluated. In this case, the nose section will be divided in a certain number of elementary sections for which it is possible to evaluate the relative CG. Following the same procedure exploited in the previous paragraph, the CG_{nose} can be evaluated combining the $CG_{structure\&TPS}$ and $CG_{avionics}$.

The information reported in Table 91 can be exploited for the CG evaluations. However, estimation refinements will be carried out by means of CAD models. They can allow to perform a first high level verification of the design requirements.

In-depth analysis: impact of hyperspectral sensor on fuselage configuration, vehicle mission and operations.

The hyperspectral sensor is one of the most currently exploited sensor with possible application on both aeronautical and space application. Besides the fact that application of this kind of sensors in hypersonic missions have not been envisaged yet, this section has the purpose of highlighting the way in which the need of integrating a component in an existing or under-development configuration has deep effect on it. In addition, depending on the characteristics of the sensor and of the configuration, proper modification of the mission and the trajectory.

The first sensors exploiting hyperspectral technology were used for remote sensing of natural environment, in particular in mineral exploration in the 1980s, highlighting, since the beginning, that the main purpose of systems based on hyperspectral technology is to identify phenomena or targets for which information about shape could be neglected and spectral data are more interesting. It is also worth to remember that hyperspectral sensors have been developed as further improvements of those equipment exploiting the multispectral technology. These enhancements were possible thanks to the main advances in focal-plane technology that allow to overcome the major disadvantages of the previous equipment. The hyperspectral sensor could be defined as a spectrometer, consisting of several advanced digital cameras able to gather electromagnetic radiation reflected by the under-observation target and to measure the energy related to each single frequency band. In particular, hyperspectral sensors are designed in order to guarantee the capability of gathering information about a few hundreds of narrow bands. This feature is the most prominent element of distinction among the various existing and under-development remote sensing instruments.

From a scientific perspective, it is obvious that the exploitation of this kind of technology relies upon the assumption that has been verified by optical studies, asserting that each material is characterized by its own spectral signature. Each pixel of the acquired picture contains the spectral information of the material. During post-processing activities, spectral signatures shall be analysed and related to a specific material and, to this purpose, a database shall be developed and test in advance. Then, data collected during the acquisition process are packaged in to a three-dimensional data structure referred to as *data cube* (Figure 193).

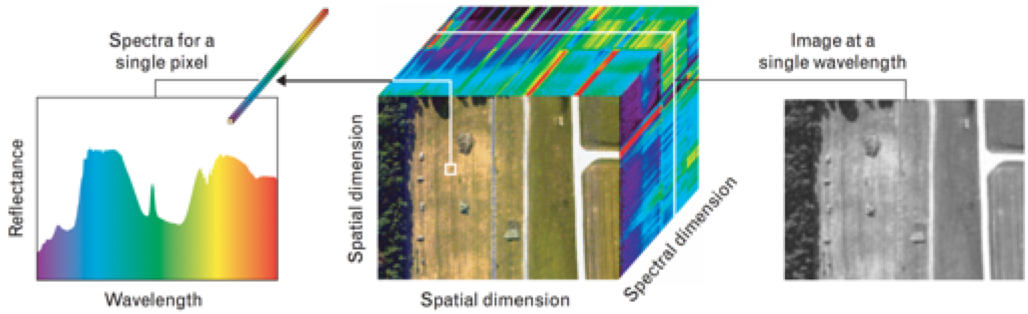


Figure 193: Data cube structure for a hyperspectral sensor

Figure 194: Example of integration of a hyperspectral sensor on-board

One of the major decision to be taken is, for example whether to use a fix mounted sensor or not, preferring installing it on a steerable turret (Figure 194). Of course, depending on this high level configuration alternative, the overall vehicle and mission can noticeably differ from the original one. For this reason, some impact analyses can be carried out to analyse the impact of such sensors on layout configuration and mission. In particular, the results shown in this section mainly refer to a case study different from the main hypersonic test case selected in Thesis, because it considers medium-sized Unmanned Aerial Vehicles (UAVs). The UAV case study was mainly developed together with Selex (currently Leonardo, Finmeccanica Company) (Fusaro, 2015) but the developed trade-off analysis for the selection of the optimal integration strategy was also in-depth analysed with the Warsaw Institute of Aviation and the Warsaw Technical University (Fusaro, 2016). Besides the difference in terms of reference vehicles and missions, the author believes that a similar approach may be applied for a future integration study of a generic sensor on the nose fuselage of a hypersonic vehicle, with the aim of optimizing performance of both the sensor itself and of the air-platform on which it will be implemented.

Moreover, this selection algorithm is here described also because it can be perfectly integrated within the multidisciplinary integrated methodology based ion SE approach described in this Thesis. Indeed, the envisaged algorithm aims at relating the stakeholder expectations not only with the vehicle architecture and performances but also to the sensor characteristics and integration aspects. Figure 195 reports an example of the first iteration of QFD (Quality Functional Deployment) tool that shows the main technical parameters that can translate in stakeholders' expectations in design variables.

	W	Vehicle architecture				Vehicle performances						Sensor characteristics					Integration	
	Importance of Stakeholders expectations	Fuselage diameter	Wing position wrt the fuselage	Empennages surface	Landing gear length	Maximum speed	Stability	Fuel consumption	Maximum range	Maximum endurance	Nominal operative altitude	Field of View	Resolution	Mass	Volume	Power consumption	Displacement wrt the CG location	Steering capability
		△	△	△	△	△	△	▽	△	△	△	△	△	▽	▽	▽	▽	△
Stakeholders expectation																		
To be exploited for a large number of monitoring scenarios	0,3					3	3		9	9	3	3	3	9	9	9	9	9
Reduced effort in data post-processing	0,2		3	3			9					3	3				9	9
Short turn around time	0,2	3			9									3	3			-3
Repeatability of the acquisition process	0,3						9	9									3	3
	1	0,6	0,6	0,6	1,8	0,9	5,4	2,7	2,7	2,7	0,9	1,5	1,5	3,3	3,3	2,7	5,4	4,8

Figure 195: Example of QFD exploitation for the selection of the optimal integration strategy for on-board systems.

It is important to notice that the technical parameters are grouped into main categories:

- vehicle architecture
    - fuselage diameter
    - wing position wrt the fuselage
    - empennages
    - landing gear type
- vehicle performances
    - maximum speed
    - aerodynamic derivatives
    - fuel consumption
    - maximum range
    - maximum endurance
    - nominal operative altitude
- Sensors characteristics
    - field of View
    - resolution
    - mass
    - volume
    - power consumption
- On-board sensor integration
    - displacement wrt the CG location
    - sensor installation type

From the exploitation of QFD, a ranking of the most affecting parameters is derived. Then, in order to compare the three different airborne platforms, these parameters should be combined and Figures of Merit obtained.

The analysis reveals that stability related characteristics (aerodynamic derivatives, relative distances of the sensor CG and aircraft CG) and mass and volume budgets are the design variables to be properly taken into account during the design process, in order to maximize the identified stakeholders' expectations. Thus, it is clear that conceptual design tools like CAD or high level CFD should be exploited in order to obtain a numerical evaluation useful for a trade-off. Moreover, technical performances such as fuel consumption and endurance should also be inserted as parameters of the Figures of Merit. The following table contains a list of the elicited Figures of Merits, related description and suggestions for their numerical evaluations in conceptual and preliminary design phases.

Table 93: Weighting strategy for the stakeholder expectations

Stakeholders Expectation	***Importance***
To be exploited for a large number of monitoring scenarios	High (0.3)
Reduced effort in post-data processing	Moderate (0.2)
Short turn-around time	Moderate (0.2)
Repeatability of the acquisition process	High (0.3)

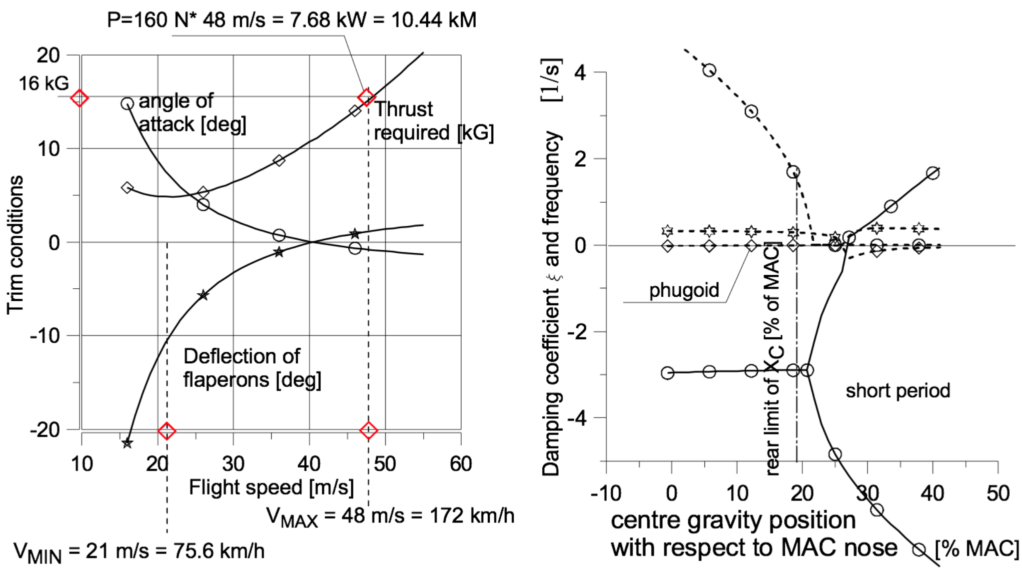


Figure 196: Stability evaluations for the Samonit UAV configuration.

It is clear that in order to estimate the parameters that are not available during the conceptual design phase, different analyses had been carried out. In particular, static and dynamic stability has to be taken into account properly (Figure 196).

Table 94: Description of the FoMs used for the trade-off

Figure of Merit	*Description*	*How to evaluate parameters*
$\frac{\boldsymbol{M_{payload} \cdot E_{max}}}{\boldsymbol{MTOM}}$ *where* $M_{payload}$ is the payload mass in kg; E_{max} is the maximum endurance in hour; $MTOM$ is the Maximum Take Off Mass.	*Unit Productivity* is a Figure of Merit that allows estimating which is the capability of the platform to host heavy payload and flying for long time duration. In this formulation, both the capabilities have the same importance but it could be sufficient to insert other weighting factors to favour one or the other.	All the parameters in the formula could be estimated or assumed in conceptual design or directly extrapolated from datasheet.
$\frac{\boldsymbol{R_{max}}}{\left(\Delta_{CG_{instal}} * \Delta_{CG_{fuel}}\right) * \boldsymbol{m_{fuel}}}$ *where* R_{max} is the maximum kilometric range; $\Delta_{CG_{instal}}$ is the relative distance among the CG of the installed sensor and the aircraft CG at the beginning of the mission; $\Delta_{CG_{fuel}}$ measures the displacement of the CG wrt its initial position (mainly due to fuel consumption during the mission); m_{fuel} is the specific fuel consumption of the installed engines.	This Figure of Merit measures the stability of the integrated system (aircraft platform and sensor) from both a static and a dynamic standpoint. Indeed, it contains a variable that measures the CGs displacement due to the installation and to the fuel consumption during the mission. In particular, this last parameter, it is strictly related to the fuel tanks location, design and fuel exploitation strategy.	The parameters inserted in the formula are strictly related to the fuel consumption that can should be iteratively evaluated exploiting simulation codes. In particular, proper tool has been created on Matlab® platform by the authors of Politecnico di Torino and reported in (Fusaro, 2015)
$\frac{\boldsymbol{T_{acquisition}}}{\boldsymbol{T_{mission}}}$ *where* $T_{acquisition}$ is the maximum useful time the sensor could be used in acquisition mode during a reference mission. $T_{mission}$ is the reference mission time duration.	This Figure of Merit provides an idea of the acquisition time and maximum endurance.	The duration of the acquisition time is strictly related to the interactions of the aircraft performances and the mission profile. Thus, simulation should be preferred.

Moreover, the evaluation of the total duration of data acquisition process during a mission has been possible exploiting an ad-hoc built-in tool developed within Politecnico di Torino (Fusaro, 2015) that allows to simulate a teledetection mission, hypothesizing a fixed mounted sensor and taking into account different flight plans. In particular, considering the results presented in (Fusaro, 2015) in order to optimize the data acquisition process, minimizing overlapping between, a sensor record and the following and minimizing the fuel consumption, a best endurance flight plan performed at fixed altitude has been selected as reference. Moreover, in case terrain profile will be considered, proper corrections to the flight plan or suggestions for the post-processing analysis will be derived by simulations carried out exploiting commercial tools like STK (Systems ToolKit) (Figure 197) or ASTOS (AeroSpace Trajectory Optimization Software).

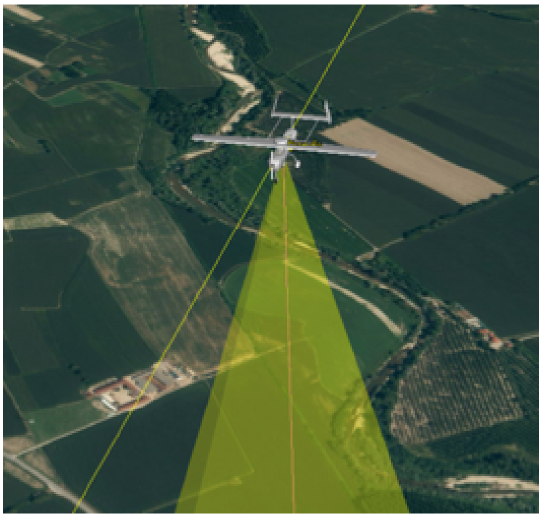


Figure 197: Example of simulation in STK®

Table 95: FoM estimation for the case study

Figure of Merit	*Parameter*	*Falco*	*Samonit 1*	*Samonit 3*
FoM 1	Payload Mass [kg]	70	20	
	Endurance [h]	14	10	
	Maximum Take Off Weight [kg]	490	80	
	Estimation of the FoM	**2,00**	**2,50**	
FoM 2	Maximum Range km (estimated)	2500	900	
	CG_sensor, % of MAC, (estimated)	-180	-100 (Sphyder® sensor only)	-130 (Sphy sensor
	CG_aircraft, % of MAC, (estimated)	-170	14	
	CG_fuel_start, % of MAC, (estimated)	-178	12	
	CG_fuel_end, % of MAC, (estimated)	-180	14	
	Fuel mass consumed, kg (estimated)	100	11	
	Estimation of the FoM	**1,25**	**0,36**	
FoM 3	Total duration of the acquisition process, [h] (estimated)	10,5	8	
	Endurance, [h]	14	10	
	Estimation of the FoM	**0,75**	**0,80**	

Table 95 shows the numerical evaluation of the Figures of Merit for the selected reference case. Considering a fourth Figure of Merit consisting in the weighted sum of the other three, the bar chart in Figure 198 shows that Samonit 3 appears to be the best solution with the only exeption of the case in which the stakeholders decide to strongly push FoM 3, giving equal importance to FoM 1 and FoM 2. The results graphically summarized in Figure 198 show that Samonit 3 is a solution robust to the weighting factors changes.

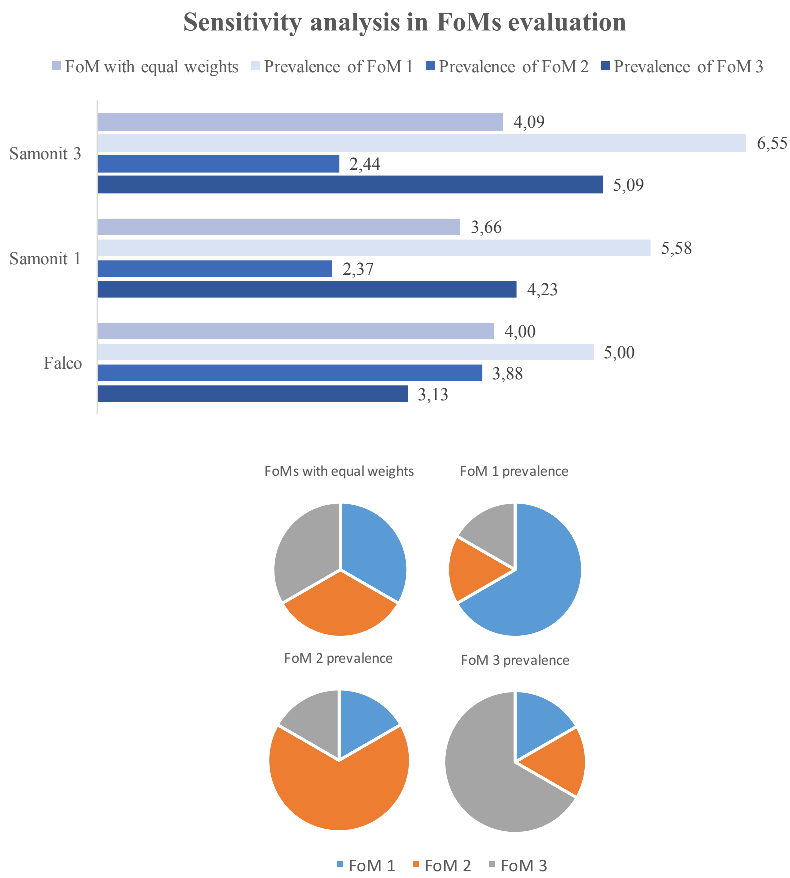


Figure 198: Sensitivity Analysis in support to the Trade-off

In-depth analysis: impact of air-intakes

The presence of air-breathing engines forces the designers to provide proper air-intakes allowing the propulsion system to be fed with fresh air all along those mission phases in which the vehicle is supposed to powered in an air-breathing mode. As it will be detailed at the end of the chapter, different possible configurations may be envisaged to integrate it in the layout, with the aim of reducing the impact on aerodynamics and aero-thermodynamics. However, in this subsection, some futuristic solutions of inlet integration in the vehicle nose are reported, giving also some useful guidelines to high level sizing. Figure 199 and Figure 200 show the way in which inlet can be integrated within the overall configuration with a noticeable impact. In addition, it is also possible to appreciate the evolution from LAPCAT MR1 and LAPCAT MR2 mainly due to the advancements in the propulsion system that allows a dramatic reduction of the inlet area due to possibility of installing a new concept of propulsion subsystem combining together ATR and DMR.

It is extremely important that the airflow into and inside the inlet duct slows down in a manner that, as velocity is reduced, the static pressure is increased. To do it there are two theoretically possible approaches:

- Expanding the cross-sectional area of the duct from the front to the back, raising up the static pressure, keeping constant the total pressure (preferable).
- Exploiting skin friction along the sides of the inlet duct. This is not considered an efficient way, because the static pressure is not raised.

It is clear that the type of geometry of the inlet and inlet duct will determine the pressure loss and distortion of the flow of the air supplied to the engine, which will affect the installed thrust and fuel consumption.

Figure 199: LAPCAT A2

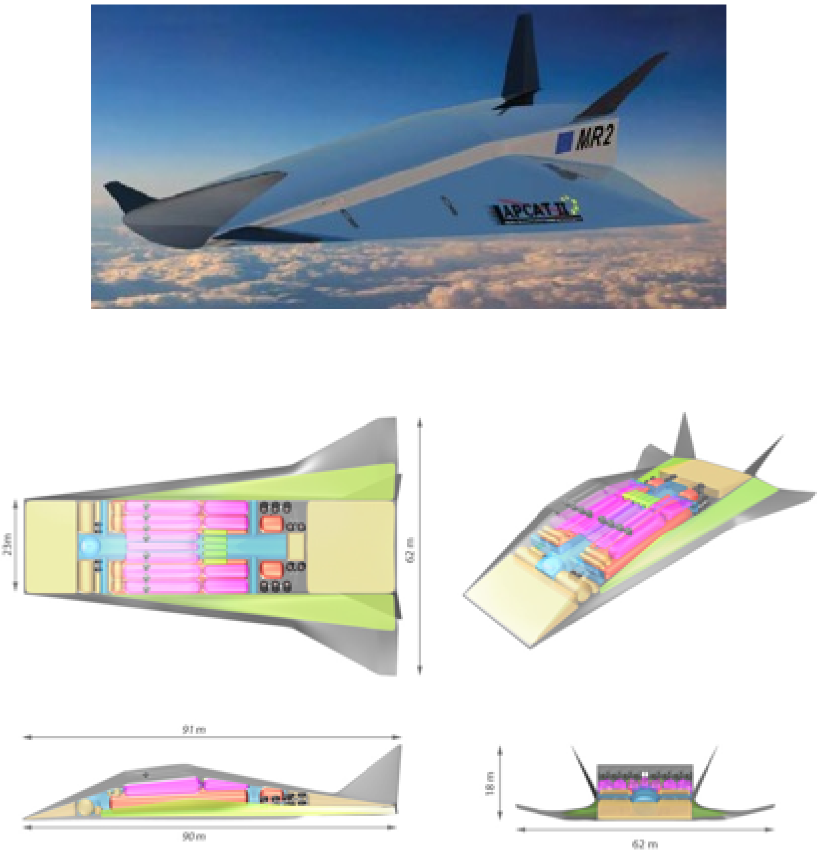


Figure 200: LAPCAT MR2

In order to have the possibility of performing a preliminary intake capture area sizing simply knowing the flight Mach Number and the mass flow, the author starts from an existing evaluation proposed by (Raymer, 2012) whose validity was confined at Mach 3. Considering that the modern hypersonic vehicle will exploit air-breathing technologies up to Mach 8, the curve has been enlarged and verified with the help of other points derived by currently under development studies.

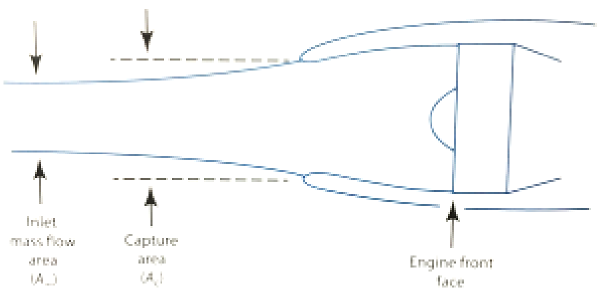


Figure 201: Generic inlet geometry

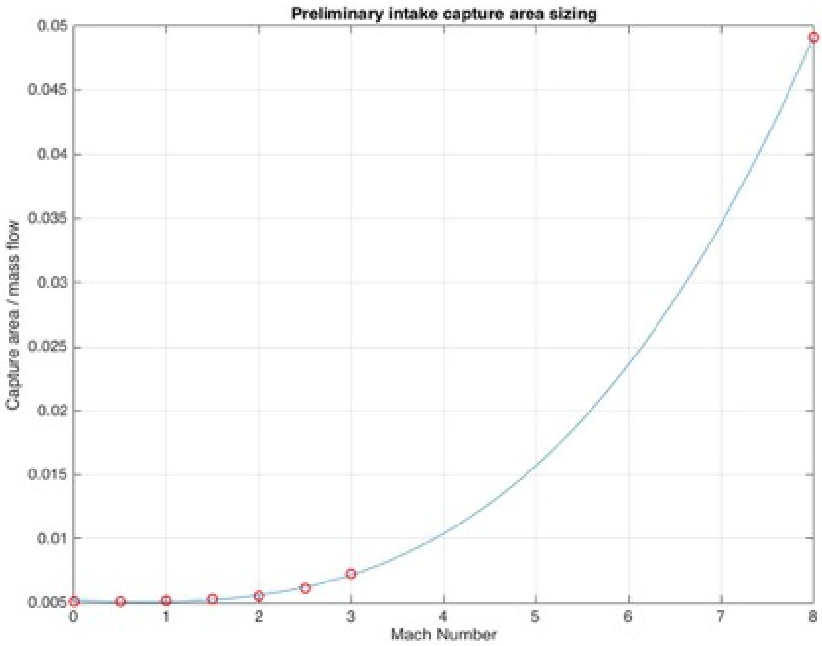


Figure 202: Preliminary intake capture area sizing

Considering Figure 201 it can be noticed that the most important design parameter is the inlet ***cross-sectional area***. It is in strict relationship with the ***capture area***, that is the measure of the cross-sectional area of the inlet front face, into the flow direction to the front-most part of the lip.

Taking a look the numbers reported in Table 96, it is possible to notice that the trends reported by Raymer, and here proposed in an enlarged version (Figure 202), the capture areas required for Mach greater than 5 are enormous and can lead to layout anomaly like the one presented in Figure 199. This is the reason why different currently under development activities are focusing on the optimization of propulsion system, aiming at reducing the inlet area without losing propulsive performances. In addition, many works are currently aiming at integrating together different propulsive types, creating single propulsive units with different operating modes, enhancing the possibility of integrating the propulsive system within the vehicle configuration.

Summarizing, it is clear that in case the engine inlet shall be placed on the nose of the vehicle, the overall configuration dramatically varies in order to accommodate the air-intake. However, besides the fact that this configuration are currently under evaluation by some important players in the field of hypersonic, the author, decides to deal with them as special configuration and for this reason, they have not been presented before. This is mainly due to the fact that this thesis aims at suggesting proper guidelines to develop innovative transportation systems to be not only more competitive but also more environmental friendly (mainly less consuming) and even safer, in order to increment the public consensus and paving the way to innovation to serve as regular flight services. But the presence of so wide inlets in the nose prevents the designer from adopting a more traditional aircraft-like layout, with the impossibility of clearly separating the area in which crew and passengers are hosted with respect to the room designed to host propellant, with a serious risk of explosion to be mitigated. However, in a futuristic approach, these very innovative configurations in which the crew compartment is no more fixed in the front fuselage, there would be the possibility of a further integration of the air inlet within the airframe.

Table 96: Air-inlet sizing for different hypersonic vehicle configurations (Steelant, 2015)

Vehicle	***Onera TBCC***	***MBDA Mk5.1***	***ESA MR2.4***
Air inlet:			
Type	2D: 2 ramps	2D axi: 2 ramps	3D streamtraced: 1 ramp + isotropic compression
Capture area (m2)	226	221	40
Air mass flowrate Mach 8 (kg/s)	4600	4500	1340
Inlet entrance section (m2)	25	50	9
External contraction ratio	8.9	4.4	4.4
Combustor entrance section (m2)	20	24	5
Internal contraction ratio	1.3	2.1	1.9
Global contraction ratio	12	9	8
Combustion chamber:			
Shape	2D rectangular	2D annular (220°)	2D elliptical
Height (m)	1	0.65	1.5
Width (m) - or perimeter	19.6	36	4
Length (m)	1.5	2.6	6

6.4.1.3 Cockpit sizing

Fuselage Cockpit Section Sizing Algorithm

Following the same rigorous approach applied for the nose sizing, also in this case, the first activity consists in understanding the list of requirements guiding the designer. Considering Table 87, reported at the beginning of this chapter, it is possible to notice that the sizing will be affected by several requirements, mainly related to the areas of aerodynamics, structure, logistics and operations (Table 98 and Figure 203).

Considering the driving requirements for the cockpit design, the activity flow proposed in Figure 204 can be followed. The process will start with the hypothesis about cockpit length that can be estimated considering the statistical population. Depending on the specific mission the vehicle shall perform, the reference statistical population may differ implying some differences in the numerical estimations. In addition, the following references (Sadraey, 2012), (Raymer, 2012), can be considered in order to have a more precise idea of the possible dimensions of the cockpit for different classes of vehicles. Eventually, it is important to notice that the actual sizing of the cockpit will be refined in the following design stages, when the specific set of avionic equipment to be installed in cabin will be identified. Ergonomic considerations may also be included to guide the designer. In addition, it can be convenient to consider this section to have a regular circular section shape, whose diameter can be considered equal to the one evaluated for the ending section of the nose part. By the way, it is clear that a check should be performed in order to understand whether or not the cockpit instruments could fit in the hypothesized area. Otherwise, a further iteration shall be carried out, and a different value for the cockpit section diameter may be identified. At the end of the sizing process, the respect of the Mach Cone envelope shall be carried out, in order to fulfil the aerodynamic and structural constraints.

It has to be noticed that in these first phases, i.e. feasibility study and conceptual design, each section will be considered separately. However, a layout harmonization process will be carried out at the end of this chapter, before moving to the integration of the fuselage with the rest of the transportation system. However, the optimal shape will be developed by means of ad-hoc aerodynamic and aerothermodynamics study.

Table 97: Requirements impact on fuselage cockpit section.

		Requirements impacting on fuselege nose section design	*Impacting Drivers*	*Impacted design parameters*	*Comments*
Comfort and Safety	*The fuselage shall safely accommodate crew members*				
	The fuselage shall safely accommodate passengers				
	The fuselage shall guarantee proper room allowing passengers to experience microgravity				
	The fuselage shall guarantee proper room allowing scientists to carry out experiments.				
	The fuselage shall guarantee a proper view of the Earth.				
Aerodynamic	*The fuselage shall generate the lowest possible drag.*		• Mach Number • Layout configuration	L_cockpit = f(μ; k_nose); D_cockpit = f(μ)	The drag produced by the fuselage cockpit section can be minimized scontained in the Mach cone, whose aperture is defined by μ = asin(1/Mach). In addition, a proper shape of the
	The fuselage heat loads shall be minimized				
	The fuselage wetted area shall be minimized.		Layout configuration	D_cockpit	external surface shall be minimized in order to avoid a drag increase.
Structure	*The fuselage weight shall be minimized.*		• Avionic equipment • Material • Cockpit section wet surface • TPS thickness and material • Primary	m_cockpit= f(S_cockpit; t_wall; m_avionic; d_material)	The mass of the cockpit fuselage section can be evaluated considering its main geometrical features, the wall thickness and the asociated materials. Moreover, the presence of avionic equipment, especially of displays, shall be
	The fuselage shall sustain the structural loads all along the flight profile.		Structural loads profile	m_cockpit = f(t_wall)	The cockpit, as well as all the different parts and components of the vehicle, shall sustain the loads all along the mission profile. To this purpose, proper
	The fuselage shape shall be as symmetric as possible.		Layot configuration	Nose section shape	The nose section shal be as symmmetric as possible, meaning that the center of the section shall be as close as possible to vehicle longitudinal axis. This will not numerically impact on nose length or
	The fuselage shall accommodate landing gear.				
	The fuselage shall accommodate propulsion systems.				
	The fuselage shall accommodate avionic subsystems		Presence of avionic apparati to be installed in the fuselage nose section.	V_cockpit = f(V_avionic); L_cockpit = f(L_avionic; D_cockpit = f(D_avionic)	The volume available in cockpit area shall be sufficiently wide to host the avionic equipmens allowing guidance and control of the vehicle. For this reasone, the avionic equipment defines
	The fuselage shall accommodate propellant subsystem				
	The internal arrangement shall guarantee the proper centre of gravity location.		Integration of avionic equipment	CG_cockpit = f(m_avionic)	The CG position of the cockpit section is mainly dependent from the integration of the
	The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.		• Mission profile • Emergency scenarios	Cockpit section shape and integration	In case of emergency, the possibility of detaching part of the fuselage shall be envisaged, in order to diminish the risk of loss of lives. In case maneuvrability should be guaranteed to the Escape
	The fuselage shall properly accommodate cargo				
Logistics and Operations	*The fuselage shall ease loading and unloading*				
	The fuselage shall guarantee proper airworthiness characteristics.		Regulations	Cockpit section shape (k_nose)	The shape of the nose fuselage section shall be properly hypothesized considering exiting
	The fuselage shall be properly designed in order to allow visibility to the pilots.		• Mission profile	α_nose = f(α_approach; V_approach) Pilot window sizing	The overnose vision shall be guaranteed. This design parameter can be evaluated on the bases of mission trajectory data such as the ange of attack (α_approach) and the speed during approch phase

Table 98: Requirements allocation on the main fuselage cockpit section design parameters.

	Cockpit Length	Cockpit Diameter	Cockpit Volume	Cockpit Mass	Cockpit CG
The fuselage shall safely accommodate crew					
The fuselage shall safely accommodate passengers					
The fuselage shall guarantee proper room allowing passengers to experience microgravity					
The fuselage shall guarantee proper room allowing scientists to carry out experiments.					
The fuselage shall guarantee a proper view of the					
The fuselage shall generate the lowest possible					
The fuselage heat loads shall be minimized					
The fuselage wetted area shall be minimized.					
The fuselage weight shall be minimized.					
The fuselage shall sustain the structural loads all along the flight profile.					
The fuselage shape shall be as symmetric as					
The fuselage shall accommodate propulsion					
The fuselage shall accommodate avionic					
The fuselage shall accommodate propellant					
The internal arrangement shall guarantee the proper centre of gravity location.					
The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.					
The fuselage shall properly accommodate cargo					
The fuselage shall ease loading and unloading					
The fuselage shall guarantee proper airworthiness characteristics.					
The fuselage shall be properly designed in order to allow visibility to the pilots.					

	CockpitLength	CockpitDiameter	CockpitVolume	CockpitMass	CockpitCG
FR33	CockpitLength	CockpitDiameter			
FR34					
FR35		CockpitDiameter			
FR36				CockpitMass	
FR37				CockpitMass	
FR38	CockpitLength	CockpitDiameter	CockpitVolume		
FR39					
FR40	CockpitLength	CockpitDiameter	CockpitVolume		
FR41					
FR42					CockpitCG
FR43	CockpitLength	CockpitDiameter	CockpitVolume		
FR44					
FR45					
FR46	CockpitLength	CockpitDiameter	CockpitVolume		
FR47		CockpitDiameter			

Figure 203: MBSE implementation of requirements allocation on the main fuselage cockpit section design parameters.

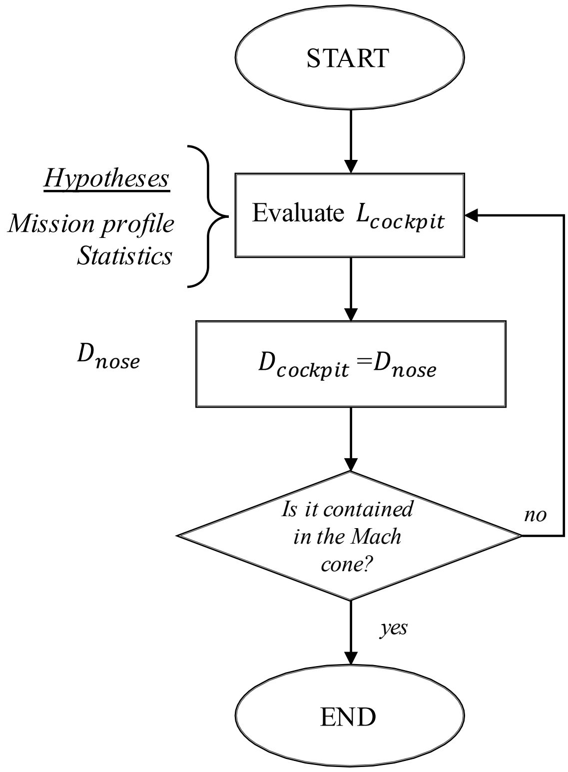


Figure 204: Flow-chart summarizing the sizing algorithm suggested for the fuselage cockpit section.

Weight and Balance estimation for the fuselage cockpit section

The overall mass of the cockpit fuselage section can be evaluated taking into account the structure of this section, the material and the presence of avionic bays. Additional design factor will be used in order to consider the mass increment due to the integration of avionics in the available volume. In addition, the presence of glass surfaces may be taken into account as well as the additional mass due to the required strengthens to guarantee the robustness of the section.

$$m_{cockpit} = \sum_{i=1}^{n_s} \left(S_{wet_i} \cdot t_i \cdot \rho_i \right) + \sum_{j=1}^{n_e} m_{avionic\ bay_j} + k_{si} + k_{str}$$

where n_s is the number of sections with different materials

S_{wet} is the wet surface of each ith-esim section;

t is the wall thickness of each ith-esim section;

ρ_i is the density of each ith-esim section;

n_e is the number of avionic equipment to be installed in the fuselage nose section;

k_{si} is the coefficient taking into account additional mass due to systems integration.

k_{str} is the coefficient taking into account the structural reinforcements required to be installed to support the structure discontinuity.

In this case, the CG relative to this part of the fuselage it's ease to be estimated thanks to the fact that this section has simpler geometrical characteristics and a part from the structure itself, the avionic equipment are the only other components to be considered.

6.4.1.4 Crew Compartment Length estimation

Fuselage Crew Compartment Section Sizing Algorithm

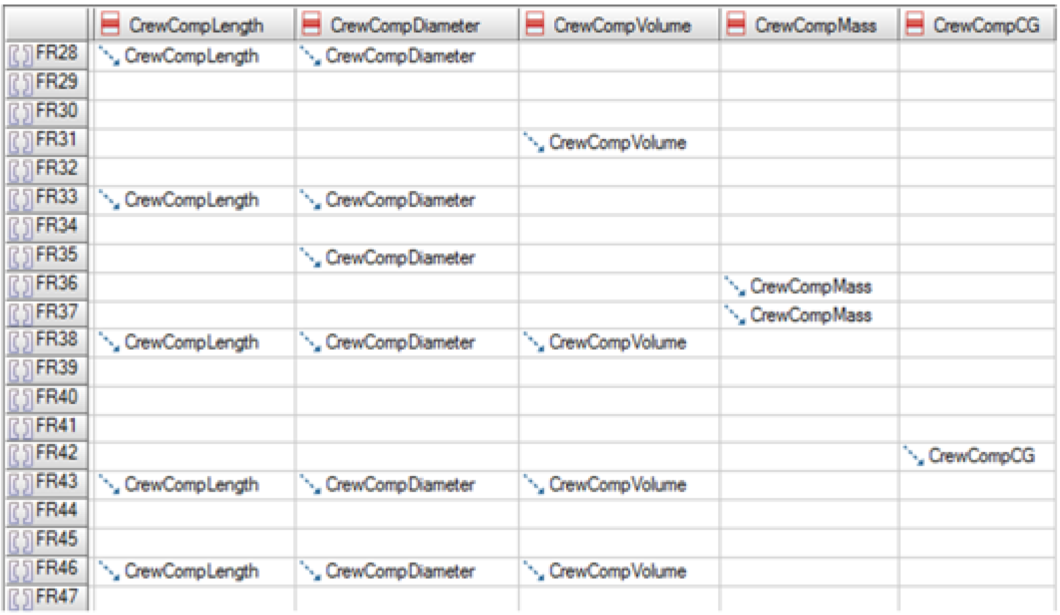
Following the same SE based approach already presented and discussed in the previous sections, the basis for the definition of a proper sizing algorithm for the fuselage Crew Compartment section has been established.

Table 99: Requirements impact on fuselage crew compartment section.

		Requirements impacting on fuselege nose section	Impacting Drivers	Impacted design parameters	Comments
Comfort and Safety	*The fuselage shall safely accommodate crew members*		n_pilot n_flight attendant	D_CrewComp = f(n_crew) L_CrewComp = f(n_crew)	Considering that the primary aim of the crew compartment is to safely accomodate pilots and flight attendants, they become the major drivers for the sizing process.
	The fuselage shall safely accommodate passengers				
	The fuselage shall guarantee proper room allowing passengers to experience microgravity				
	The fuselage shall guarantee proper room allowing scientists to carry out experiments.		scientific payload n_scientists	V_CrewComp = f(V_payload)	In case, during the mission, some scientific experiments should be carried out, the crew compartment shall guarantee proper room for both scientists and payload.
	The fuselage shall guarantee a proper view of the Earth.				
Aerodynamic	*The fuselage shall generate the lowest possible drag.*		• Mach Number • Layout configuration	D_CrewComp = f(n_crew) L_CrewComp = f(n_crew)	The drag produced by the fuselage crew compartment section can be minimized scontained in the Mach cone, whose aperture is defined by μ = asin(1/Mach).
	The fuselage heat loads shall be minimized				
	The fuselage wetted area shall be minimized.		Layout configuration	D_CrewComp	The fuselage crew compartment section external surface shall be minimized in order to avoid a drag increase.
Structure	*The fuselage weight shall be minimized.*		• Scientific payload • Scientific payload • Material • Wet surface • TPS thickness and material • Primary structure thickness and material • Furnishing	m_CrewComp	The mass of the Crew Compartment fuselage section can be evaluated considering its main geometrical features, the wall thickness and the asociated materials. Moreover, the presence of scientific payloads, shall be included in the mass estimation, as well as, the presence of pilot and scientists. Flight attendants are not included in this evaluation, but considering they spend the majority of the time in the Passenger
	The fuselage shall sustain the structural loads all along the flight profile.		Structural loads profile	m_CrewCompartment = f(t_wall)	The Crew Compartment, as well as all the different parts and components of the vehicle, shall sustain the loads all along the mission profile. To this purpose, proper material and thicknesses of the sections must
	The fuselage shape shall be as symmetric as possible.		Layot configuration	Crew Compartment section shape	The CRew Compartment section shal be as symmmetric as possible, meaning that the center of the section shall be as close as possible to vehicle longitudinal axis. This will not numerically impact on nose length or diameter directly, but it has a primary
	The fuselage shall accommodate landing gear.				
	The fuselage shall accommodate propulsion systems.				
	The fuselage shall accommodate avionic subsystems				
	The fuselage shall accommodate propellant subsystem				
	The internal arrangement shall guarantee the proper centre of gravity location.		• Integration of furnishing and scientific payload • m_pilot • m_scientists	CG_Crew Compartment= f(m_structure, m_furnishing, m_payload, m_crew and m_scient)	The CG position of the cockpit section is mainly dependent from the integration of the avionic equipment installed inside.
	The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.		• Mission profile • Emergency scenarios	Crew Compartment section shape and integration	In case of emergency, the possibility of detaching part of the fuselage shall be envisaged, in order to diminish the risk of
	The fuselage shall properly accommodate cargo				
Logistics and Operations	*The fuselage shall ease loading and unloading operations*				
	The fuselage shall guarantee proper airworthiness characteristics.		Regulations	Crew compartment section shape	The shape of the nose fuselage section shall be properly hypothesized considering exiting airworthiness regulations
	The fuselage shall be properly designed in order to allow visibility to the pilots.				

Table 100: Requirements allocation on the main fuselage crew compartment section design parameters.

	Crew Compartment Length	Crew Compartment Diameter	Crew Compartment Volume	Crew Compartment Mass	Crew Compartment CG
The fuselage shall safely accommodate crew members					
The fuselage shall safely accommodate passengers					
The fuselage shall guarantee proper room allowing passengers to experience microgravity					
The fuselage shall guarantee proper room allowing scientists to carry out experiments.					
The fuselage shall guarantee a proper view of the Earth					
The fuselage shall generate the lowest possible drag.					
The fuselage heat loads shall be minimized					
The fuselage wetted area shall be minimized.					
The fuselage weight shall be minimized.					
The fuselage shall sustain the structural loads all along the flight profile.					
The fuselage shape shall be as symmetric as possible.					
The fuselage shall accommodate propulsion systems.					
The fuselage shall accommodate avionic subsystems					
The fuselage shall accommodate propellant subsystem					
The internal arrangement shall guarantee the proper centre of gravity location.					
The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.					
The fuselage shall properly accommodate cargo					
The fuselage shall ease loading and unloading					
The fuselage shall guarantee proper airworthiness characteristics.					
The fuselage shall be properly designed in order to allow visibility to the pilots.					



	CrewCompLength	CrewCompDiameter	CrewCompVolume	CrewCompMass	CrewCompCG
FR28	CrewCompLength	CrewCompDiameter			
FR29					
FR30					
FR31			CrewCompVolume		
FR32					
FR33	CrewCompLength	CrewCompDiameter			
FR34					
FR35		CrewCompDiameter			
FR36				CrewCompMass	
FR37				CrewCompMass	
FR38	CrewCompLength	CrewCompDiameter	CrewCompVolume		
FR39					
FR40					
FR41					
FR42					CrewCompCG
FR43	CrewCompLength	CrewCompDiameter	CrewCompVolume		
FR44					
FR45					
FR46	CrewCompLength	CrewCompDiameter	CrewCompVolume		
FR47					

Figure 205: MBSE implementation of requirements allocation on the main fuselage crew compartment section design parameters.

Mathematical details for the fuselage crew compartment section sizing

In order to size the crew compartment, standards usually considered for military aircraft can be applied, especially in case of orbital, suborbital or re-entry vehicles. Complementary, in case of more conventional missions, such as the point-to-point mission, commercial aviation standards can be applied. In any case the overall crew compartment length is mainly defined by the sum of the pilot seats pitch and the possible empty space aimed at hosting additional pilot cabin support systems or simply aimed at guaranteeing proper accessibility and mobility inside. It is important to notice that in case of orbital or suborbital vehicles aimed at allowing scientific experiments during the mission, a proper empty space can be added in the rear part of the crew compartment. This innovative configuration harks back to those ones in which flight engineer role and accommodation was envisaged.

In the end, the crew compartment length ($L_{crew\ compartment}$) can be evaluated as follows:

$$L_{crew\ compartment} = l_{seat\ pitch} + l_{empty\ space}$$

where

$l_{seat\ pitch}$ is the pilot seat pitch length;

$l_{empty\ space}$ is the estimation of the empty space that the designer would add for the above mentioned reasons.

The top view of the crew compartment can be defined properly locating the pilot and flight attendants' seats on the available volume. The available volume can be easily identified simply extending the borders of the nose and cockpit sections gradually reducing their inclination with respect to the symmetry line of the longitudinal plane. In order to have a first estimation of the diameter at the crew compartment extremities, simple calculations can be performed. While the minimum value can be considered coincident with the diameter of the cockpit, the following formulation can be adopted to estimate the other extremity section diameter.

$$\left(D_{crew\ compartment}\right)_{max} = n_{pilot} \cdot w_{pilot\ seat} + (1 + k_{crew}) \cdot (b_{seat} - w_{pilot\ seat}) \cdot k_s$$

where

n_{pilot} is the number of pilots;

$w_{pilot\ seat}$ is the pilot's seat width;

b_{seat} is the distance between two pilot's seats measured from/to the seats CGs.;

k_{crew} is a parameter that allows estimating the additional space that should be considered. Considering the role of this parameter in the equation, it is clear that additional space is considered in relationship with the aisle between the seats $(b_{seat} - w_{pilot\ seat})$.

k_s is a safety factor that allows to take into account an enlargement of the fuselage diameter;

The crew compartment section layout depends on the envisaged lofting strategy that has been selected. In case of small hypersonic vehicles, with a single pilot, the configuration of typical fighter crew compartment can be adopted. As reference. Complementary, in case of bigger transportation systems, where two pilots are present, the typical configuration of modern commercial aircraft will be considered.

Cargo Section Design

In a transport aircraft aimed at accomodating passengers, a large section of the fuselage should be allocated to the cargo/luggage bay. Thus, the fuselage cross-section, at least the passenger compartment cross section may be designed in such a way that it can encompasses a sufficient volume for cargo/luggage. The free volume available to these purposes directly comes from the requests of the airliners that allows passengers checking in a certain number of luggage with a proper size. Of course, oversized as well as overweight luggage should be admitted. However, in order to increase the luggage safety during flight, reducing excessive movements, to save loading and unloading time, shortening the turn-around time, preventing delayed flight, standardized containers and pallets are used. In Figure 102, some IATA standardized containers types are reported.

The total volume of passenger cargo V_C can be estimated by the following equation:

$$V_C = n_t \cdot V_b$$

where

n_t is the number of travellers, including passengers, crew and flight attendants;

V_b is the total baggage volume of each traveller; considering values suggested in literature (Sadraey, 2012), a value of 0,146 m^3 is here suggested.

Table 101: Suggestions for standard cargo containers sizing.

Identification	Width [in. (mm)]	Height [in. (mm)]	Depth [in. (mm)]	Volume [ft^3 (m^3)]	Maximum Load [lb (kg)]
LD 1	92 (2336,8)	64 (1625,6)	60 (1524)	173	3500
LD 2	61,5 (1562,1)	64 (1625,6)	47 (1193,8)	120	2700
LD 3	79 (2006,6)	64 (1625,6)	60,4 (1534,16)	159	3500
LD 4	96 (2438,4)	64 (1625,6)	60,4 (1534,16)	-	5400
LD 5	125 (3175)	64 (1625,6)	60,4 (1534,16)	-	7000
LD 6	160 (4064)	64 (1625,6)	60,4 (1534,16)	316	7000
LD 7	125 (3175)	64 (1625,6)	80 (2032)	381	13300
LD 8	125 (3175)	64 (1625,6)	60,4 (1534,16)	243	5400
LD 9	125 (3175)	64 (1625,6)	80 (2032)	-	13300
LD 10	125 (3175)	64 (1625,6)	60,4 (1534,16)	-	7000
LD 11	125 (3175)	64 (1625,6)	60,4 (1534,16)	253	7000
LD 12	1864724,4	64 (1625,6)	88 (2235,2)	-	13300

Weight & balance considerations for the fuselage crew compartment section sizing

As far as the mass and CG evaluations are concerned, they can be easily performed similarly to what is done in the other above detailed fuselage sections, with the addition of considering pilots, and the possible presence of scientific experiments, possible racks in which they can be properly accommodated and the scientists that will operate it. Thus, the following formulation can be adopted to perform a preliminary mass estimation:

$$m_{CrewComp} = \sum_{i=1}^{n_s} \left(S_{wet_i} \cdot t_i \cdot \rho_i \right) + m_{payload} + k_{si} + n_{pilot} m_{pilot} + n_{scientists} m_{scientists}$$

where n_s is the number of sections with different materials

S_{wet} is the wet surface of each ith-esim section;

t is the wall thickness of each ith-esim section;

ρ_i is the density of each ith-esim section;

k_{si} is the coefficient taking into account additional mass due to systems integration;

$m_{payload}$ is the mass of the scientific payload;

n_{pilot} is the number of pilots to be taken into account;

m_{pilot} is the mass per pilots (it can slightly vary by genre and whether or not they are supposed to wear special suit or additional safety equipment such oxygen mask and dispensers);

$n_{scientists}$ is the number of scientists that will operate the payload during the mission;

$m_{scientists}$ is the mass per scientist (it can slightly vary by genre and whether or not they are supposed to wear special suit or additional safety equipment such oxygen mask and dispensers);

6.4.1.5 Passenger Compartment Length estimation

Fuselage Passengers Compartment Section Sizing Algorithm

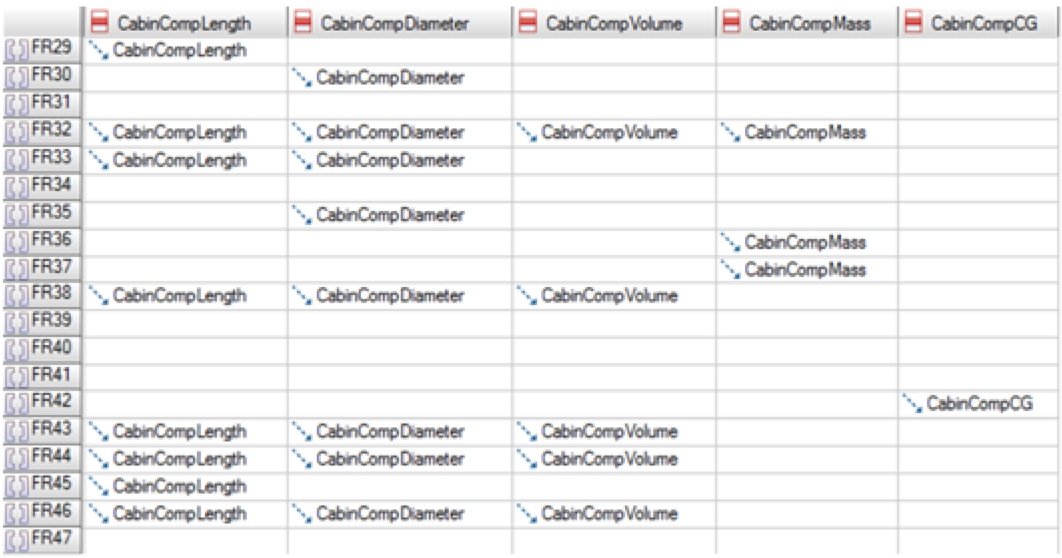
Following the same SE based approach already presented and discussed in the previous sections, the basis for the definition of a proper sizing algorithm for the fuselage Passengers Compartment section has been established.

Table 102: Requirements impact on fuselage passengers compartment section.

		Requirements impacting on fuselege	Impacting Drivers	Impacted design parameters	Comments
Comfort and Safety	*The fuselage shall safely accommodate crew members*		• n_pilot • n_flight attendant	D_CrewComp = f(n_crew) L_CrewComp = f(n_crew)	Considering that the primary aim of the crew compartment is to safely accomodate pilots and flight attendants, they become the major
	The fuselage shall safely accommodate				
	The fuselage shall guarantee proper room allowing passengers to experience microgravity				
	The fuselage shall guarantee proper room allowing scientists to carry out experiments.		• scientific payload • n_scientists	V_CrewComp = f(V_payload)	In case, during the mission, some scientific experiments should be carried out, the crew compartment shall guarantee proper room for both
	The fuselage shall guarantee a proper view of the				
Aerodynamic	*The fuselage shall generate the lowest possible drag.*		• Mach Number • Layout configuration	D_CrewComp = f(n_crew) L_CrewComp = f(n_crew)	The drag produced by the fuselage crew compartment section can be minimized scontained in the Mach cone, whose aperture is defined by µ
	The fuselage heat loads shall be minimized				
	The fuselage wetted area shall be minimized.		Layout configuration	D_CrewComp	section external surface shall be minimized in order to avoid a drag increase.
Structure	*The fuselage weight shall be minimized.*		• Scientific payload • Material • Wet surface • TPS thickness and material • Primary structure thickness and material	m_CrewComp	The mass of the Crew Compartment fuselage section can be evaluated considering its main geometrical features, the wall thickness and the asociated materials. Moreover, the presence of scientific payloads, shall be included in the mass estimation, as well as, the presence of pilot and scientists. Flight attendants are not included in this evaluation, but
	The fuselage shall sustain the structural loads all along the flight profile.		Structural loads profile	m_CrewCompartment = f(t_wall)	The Crew Compartment, as well as all the different parts and components of the vehicle, shall sustain the loads all along the mission profile. To this purpose, proper material and
	The fuselage shape shall be as symmetric as possible.		Layot configuration	Crew Compartment section shape	The Crew Compartment section shal be as symmmetric as possible, meaning that the center of the section shall be as close as possible to vehicle longitudinal axis. This will not numerically impact on nose length or
	The fuselage shall accommodate landing gear.				
	The fuselage shall accommodate propulsion				
	The fuselage shall accommodate avionic				
	The fuselage shall accommodate propellant				
	The internal arrangement shall guarantee the proper centre of gravity location.		• Integration of furnishing and scientific payload • m_pilot	CG_Crew Compartment= f(m_structure, m_furnishing, m_payload, m_crew	The CG position of the Crew Compartment section is mainly dependent from the integration of the avionic equipment installed inside.
	The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.		• Mission profile • Emergency scenarios	Crew Compartment section shape and integration	In case of emergency, the possibility of detaching part of the fuselage shall be envisaged, in order to diminish the
	The fuselage shall properly accommodate cargo				
Logistics and Operations	*The fuselage shall ease loading and unloading*				
	The fuselage shall guarantee proper airworthiness characteristics.		Regulations	Crew compartment section shape	The shape of the Crew Compartment shall be properly hypothesized considering exiting airworthiness
	The fuselage shall be properly designed in order to allow visibility to the pilots.				

Table 103: Requirements allocation on the main fuselage passengers section design parameters.

	Passengers Compartment Length	Passengers Compartment Diameter	Passengers Compartment Volume	Passengers Compartment Mass	Passengers Compartment CG
The fuselage shall safely accommodate crew					
The fuselage shall safely accommodate passengers					
The fuselage shall guarantee proper room allowing passengers to experience microgravity					
The fuselage shall guarantee proper room allowing scientists to carry out experiments.					
The fuselage shall guarantee a proper view of the					
The fuselage shall generate the lowest possible drag.					
The fuselage heat loads shall be minimized					
The fuselage wetted area shall be minimized.					
The fuselage weight shall be minimized.					
The fuselage shall sustain the structural loads all along the flight profile.					
The fuselage shape shall be as symmetric as					
The fuselage shall accommodate propulsion					
The fuselage shall accommodate avionic subsystems					
The fuselage shall accommodate propellant					
The internal arrangement shall guarantee the proper centre of gravity location.					
The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.					
The fuselage shall properly accommodate cargo					
The fuselage shall ease loading and unloading					
The fuselage shall guarantee proper airworthiness characteristics.					
The fuselage shall be properly designed in order to allow visibility to the pilots.					



	CabinCompLength	CabinCompDiameter	CabinCompVolume	CabinCompMass	CabinCompCG
FR29	CabinCompLength				
FR30		CabinCompDiameter			
FR31					
FR32	CabinCompLength	CabinCompDiameter	CabinCompVolume	CabinCompMass	
FR33	CabinCompLength	CabinCompDiameter			
FR34					
FR35		CabinCompDiameter			
FR36				CabinCompMass	
FR37				CabinCompMass	
FR38	CabinCompLength	CabinCompDiameter	CabinCompVolume		
FR39					
FR40					
FR41					
FR42					CabinCompCG
FR43	CabinCompLength	CabinCompDiameter	CabinCompVolume		
FR44	CabinCompLength	CabinCompDiameter	CabinCompVolume		
FR45	CabinCompLength				
FR46	CabinCompLength	CabinCompDiameter	CabinCompVolume		
FR47					

Figure 206: MBSE implementation of requirements allocation on the main fuselage passengers compartment section design parameters.

Mathematical details for the Fuselage Passengers Compartment Section Sizing

Considering the passengers compartment, it is possible noticing that the sizing is deeply affected by the number of passengers. Indeed, this parameter has implications on many of the other elements including the number and dimensions of the exit doors, galleys and toilets. This is also perfectly traced in Table 103, summarizing the impact of fuselage requirements on the major design parameters of the passenger compartment.

The overall passenger compartment length estimation can be carried out solving the following equations:

$$L_{pax\,comp} = \max(L_{pax_i})$$

with

$$\begin{aligned} L_{pax_i} = \sum_{j=1}^{n_{classes}} &((n_{row})_j \cdot (L_{seat\,pitch})_j + k_0 \\ &\cdot \left(n_{galley} \cdot l_{galley} + n_{toilet} \cdot l_{toilet} +\cdot\, n_{wardrobe} \cdot l_{wardrobe}\right) \\ &+ \sum_{l=1}^{n_{doors\,type}} \{(l_{door})_l \cdot (n_{door})_l\} + l_{TypeA}) \end{aligned}$$

$$W_{pax\,comp} = \sum_{j=1}^{n_{classes}} ((n_{aisle} + 1) \cdot w_{seat} + n_{aisle} \cdot w_{aisle})$$

$$H_{pax\,comp} = \sum_{i=1}^{n_{decks}} (h_{aisle}) + h_{cargo}$$

where

n_{decks} is the number of decks of the configuration;

$n_{classes}$ is the number of classes in which each i-esim deck has been organized;

n_{aisle} is the number of aisle in each j-esim class;

n_{row} is the number of rows in each j-esim class. The number of row can be identified by mean of the number of passengers per row and the number of seat abreast per each group of seats, this equation $(n_{row})_j = \frac{(n_{pax})_j}{\sum_{i=1}^{n_{aisle}+1}[(n_{seat\ abreast})_j]_i}$

k_0 is a safety design parameter

n_{galley} is the number of required galleys that is a direct function of the number of passengers and of the mission duration (see. CS 25). In addition, Considering an axial symmetric external shape with a symmetric equipment installation

- in case n_{galley} is an even number, $n_{galley} = \frac{n_{galley}}{2}$;
- otherwise $n_{galley} = floor\left(\frac{n_{galley}}{2}\right) + 1$;

l_{galley} is the length of each single galley. As reference, the reader can exploit data reported in CS 25.

n_{toilet} is the number of required toilets that is a direct function of the number of passengers (see. CS 25). In addition, Considering an axial symmetric external shape with a symmetric equipment installation

- in case n_{toilet} is an even number, $n_{toilet} = \frac{n_{toilet}}{2}$;
- otherwise $n_{toilet} = floor\left(\frac{n_{toilet}}{2}\right) + 1$;

l_{toilet} is the length of each single toilet. As reference, the reader can exploit data reported in CS 25.

$n_{wardrobe}$ is the number of required wardrobes that is a direct function of the number of passengers (see. CS 25). In addition, Considering an axial symmetric external shape with a symmetric equipment installation

- in case $n_{wardrobe}$ is an even number, $n_{toilet} = \frac{n_{wardrobe}}{2}$;
- otherwise $n_{wardrobe} = floor\left(\frac{n_{wardrobe}}{2}\right) + 1$;

$n_{doors\ type}$ is the number of different doors' type to be placed in each deck;

l_{door} is the l-th door's type length;

n_{door} is the number of doors of the l-th door's type. In addition, Considering an axial symmetric external shape with a symmetric equipment installation

- in case n_{doors} is an even number, $n_{doors} = \frac{n_{doors}}{2}$;
- otherwise $n_{doors} = floor\left(\frac{n_{doors}}{2}\right) + 1$;

l_{TypeA} is the length of a Type A door.

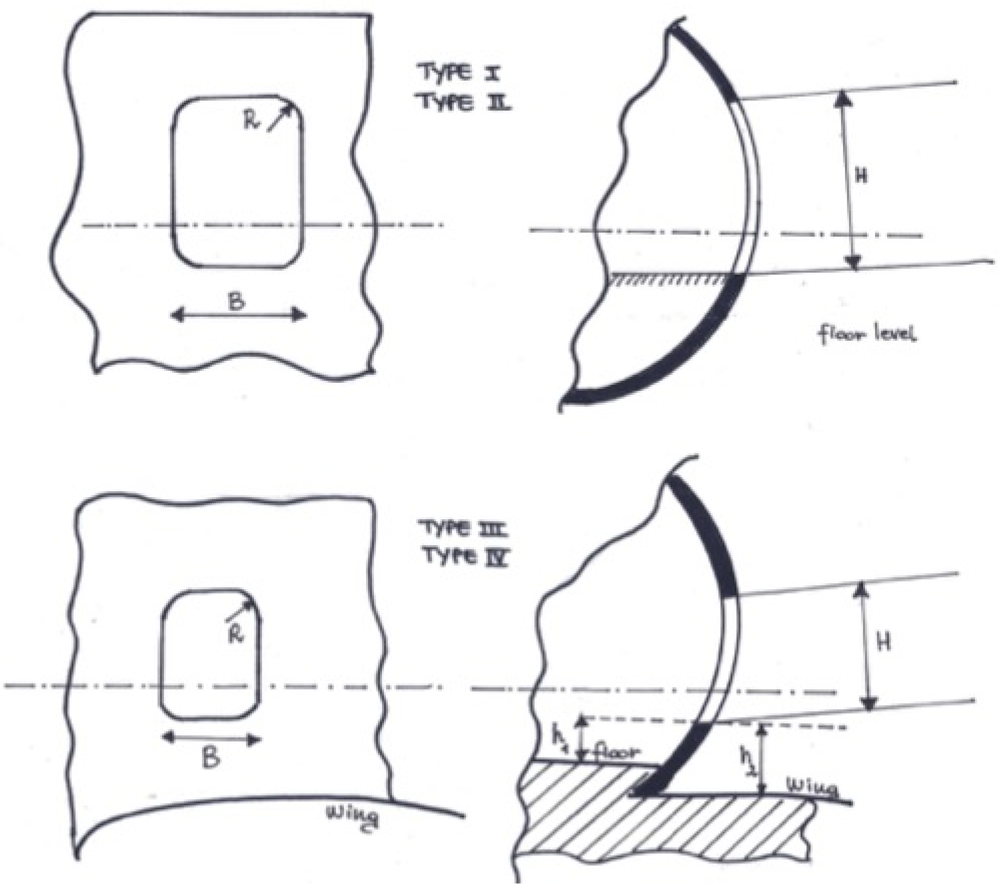


Figure 207: Sizing suggestions for Emergency Exits (for numerical values of the parameters refer to CS 25)

Weight &Balance considerations

The mass of this section has different contributions: the passengers mass, the mass of the seats and of all the furnishing and subsystems to be integrated within the passengers' compartment. In particular, the following equation tries to collect all the different contributes. However, depending on the specific case study, it may be enlarged in order to take into account additional factors.

On the other hand, noticing that the overall mass of this section is homogeneously distributed all along the length, in a first attempt, the CG of this compartment can be assumed to be located where the geometrical CG is located.

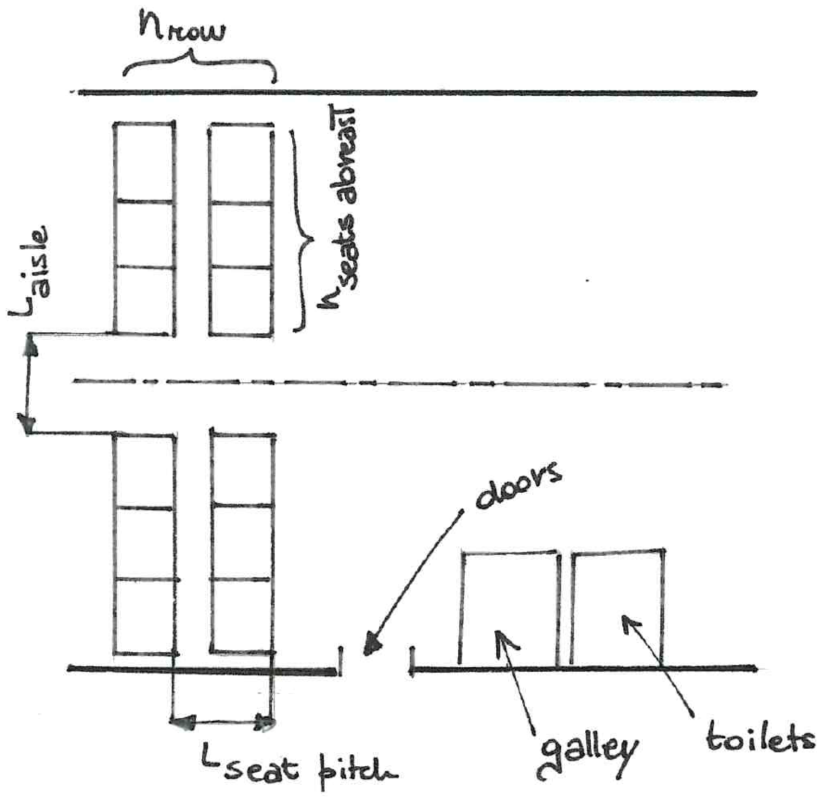


Figure 208: Sketch of a generic passengers compartment section

6.4.2 Aft-fuselage Length estimation

6.4.2.1 Aft Fuselage Section Sizing Algorithm

The aft fuselage section can be considered to be the best section in which the majority of the subsystems can be installed. In particular, it can host the propulsion and propellant subsystems, ensuring proper distance and separation from the crew and passengers compartments. Thus, the sizing of this section can be carried out simply estimating the length and diameter of engines, rockets and propellant tanks. The following table summarizes the major relationships between requirements and design parameters.

Table 104: Requirements allocation on the main aft fuselage section design parameters.

	Aft Fuselage Length	**Aft Fuselage Diameter**	**Aft Fuselage Volume**	**Aft Fuselage Mass**	**Aft Fuselage CG**
The fuselage shall safely accommodate crew members					
The fuselage shall safely accommodate passengers					
The fuselage shall guarantee proper room allowing passengers to experience microgravity					
The fuselage shall guarantee proper room allowing scientists to carry out experiments.					
The fuselage shall guarantee a proper view of the Earth.					
The fuselage shall generate the lowest possible drag.					
The fuselage heat loads shall be minimized					
The fuselage wetted area shall be minimized.					
The fuselage weight shall be minimized.					
The fuselage shall sustain the structural loads all along the flight profile.					
The fuselage shape shall be as symmetric as possible.					
The fuselage shall accommodate propulsion systems.					
The fuselage shall accommodate avionic subsystems					
The fuselage shall accommodate propellant subsystem					
The internal arrangement shall guarantee the proper centre of gravity location.					
The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.					
The fuselage shall properly accommodate cargo					
The fuselage shall ease loading and unloading operations					
The fuselage shall guarantee proper airworthiness characteristics.					
The fuselage shall be properly designed in order to allow visibility to the pilots.					

Table 105: Requirements impact on the aft fuselage section.

		Requirements impacting on fuselege	*Impacting Drivers*	*Impacted design parameters*	*Comments*
Comfort and Safety	*The fuselage shall safely accommodate crew members*				
	The fuselage shall safely accommodate passengers				
	The fuselage shall guarantee proper room allowing passengers to experience microgravity				
	The fuselage shall guarantee proper room allowing scientists to carry out experiments.				
	The fuselage shall guarantee a proper view of the				
Aerodynamic	*The fuselage shall generate the lowest possible drag.*		• Mach Number • Layout	D_Aft L_Aft	The drag produced by the fuselage passenger compartment section can be
	The fuselage heat loads shall be minimized				
	The fuselage wetted area shall be minimized.		Layout configuration	D_PaxComp	The aft fuselage section external surface shall be minimized in order to avoid a drag increase.
Structure	*The fuselage weight shall be minimized.*		• Material • Wet surface • TPS thickness and material • Primary structure thickness and	m_Aft = f(m_struct, m_sys)	The mass of the Aft fuselage section can be evaluated considering its main geometrical features, the wall thickness and the asociated materials. In addition, the presence of subsystems to be integrated inside shall be considered.
	The fuselage shall sustain the structural loads all along the flight profile.		Structural loads profile	m_Aft= f(t_wall)	The aft fuselage section as well as all the different parts and components of the vehicle, shall sustain the loads all along the mission profile. To this purpose, proper material and thicknesses
	The fuselage shape shall be as symmetric as possible.		Layout configuration	Crew Compartment section shape	The Passengers Compartment section shal be as symmmetric as possible, meaning that the center of the section shall be as close as possible to vehicle longitudinal axis. This will not numerically impact on nose
	The fuselage shall accommodate propulsion systems.		• Engine Type • Engine Technology • Thrust	L_Aft = f (L_engine) D_Aft = f (D_engine)	Length and Diameter of the aft fuselage are strictly related to the sizing of the propulsiove system hosted inside.
	The fuselage shall accommodate avionic subsystems				
	The fuselage shall accommodate propellant subsystem				
	The internal arrangement shall guarantee the proper centre of gravity location.		• Integration of subsystems • m_propulsion • m_tank_dry	CG_Aft = f(m_structure, m_propulsion, m_propellant)	The CG position of the aft fuselage section is mainly dependent from the integration of the propulsion and propellant systems within the available
	The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.				
	The fuselage shall properly accommodate cargo				
Logistics and Operations	*The fuselage shall ease loading and unloading*				
	The fuselage shall guarantee proper airworthiness characteristics.		Regulations	Aft fuselage section shape	The shape of the aft fuselage section shall be properly hypothesized considering exiting
	The fuselage shall be properly designed in order to allow visibility to the pilots.				

	SystemsCompLength	SystemsCompDiameter	SystemsCompVolume	SystemsCompMass	SystemsCompCG
FR33	SystemsCompLength	SystemsCompDiameter			
FR34					
FR35		SystemsCompDiameter			
FR36				SystemsCompMass	
FR37				SystemsCompMass	
FR38	SystemsCompLength	SystemsCompDiameter	SystemsCompVolume		
FR39	SystemsCompLength	SystemsCompDiameter			
FR40					
FR41					
FR42					SystemsCompCG
FR43					
FR44					
FR45					
FR46	SystemsCompLength	SystemsCompDiameter	SystemsCompVolume		
FR47					

Figure 209: MBSE implementation of requirements allocation on the main aft fuselage section design parameters.

Whatever propulsion technology will be selected, both in the air-breathing and in the non-air-breathing domain, two different design processes may be followed:

1. In case a suitable existing hardware perfectly fitting with performance requirements, the dimension of the selected component will be taken as reference and the aft fuselage properly sized to accommodate it.
2. On the other case, whether existing components cannot perfectly match the design points, the rubber engine strategy may be adopted. The most similar component, in terms of required performance or simply in terms of technology, should be selected as reference engine. Then, as it is highlighted in Figure 211, proper scaling laws will be applied in order to derive the best customized solution.

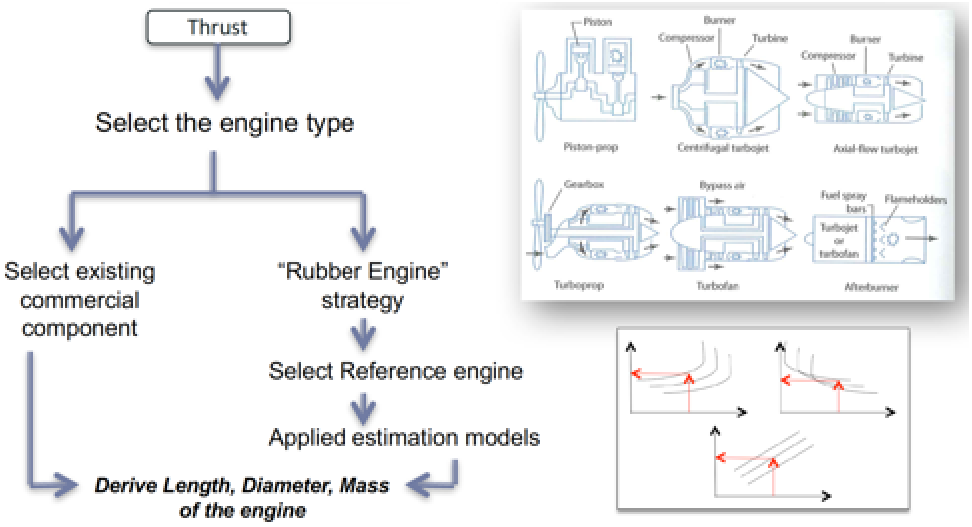


Figure 210: Sizing flow-chart for a generic air-breathing engine

$$SF = \frac{T_{required}}{T_{actual}} \Rightarrow \begin{cases} L = L_{actual}(SF)^{0.4} \\ D = D_{actual}(SF)^{0.5} \\ W = W_{actual}(SF)^{1.1} \end{cases}$$

Non-afterburnirg engines

$$W = 14.7T^{1.1}e^{(-0.045\,BPR)}$$
$$L = 0.49T^{0.4}M^{0.2}$$
$$D = 0.15T^{0.5}e^{(0.04\,BPR)}$$

Afterburning engines

$$W = 11.1T^{1.1}M^{0.25}e^{(-0.81\,BPR)}$$
$$L = 0.68T^{0.4}M^{0.2}$$
$$D = 0.11T^{0.5}e^{(0.04\,BPR)}$$

Figure 211: Example of rubber engine strategy for the preliminary sizing of a generic air-breathing engine.

It is crystal clear that the level of confidence of the estimations will be impacted by the engines database from which reference components can be selected and from which the scaling factors and the parameters of each single equation can be properly tuned.

A far as the fuel and propellant mass is concerned, in both cases, the major inputs directly comes from the high level estimations, previously performed, and from the latest mission outputs. From masses, it is possible to derive the volumes required to host fuel, propellant and oxidizer knowing the densities.

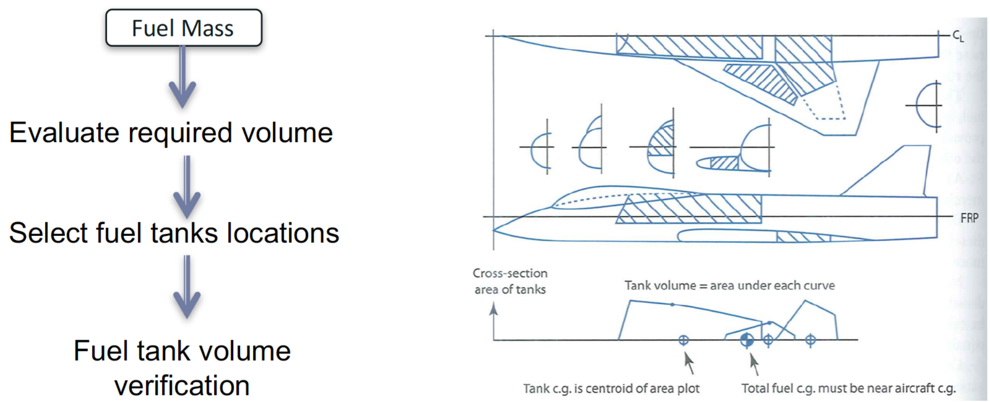


Figure 212: Flow-chart for Fuel tanks design

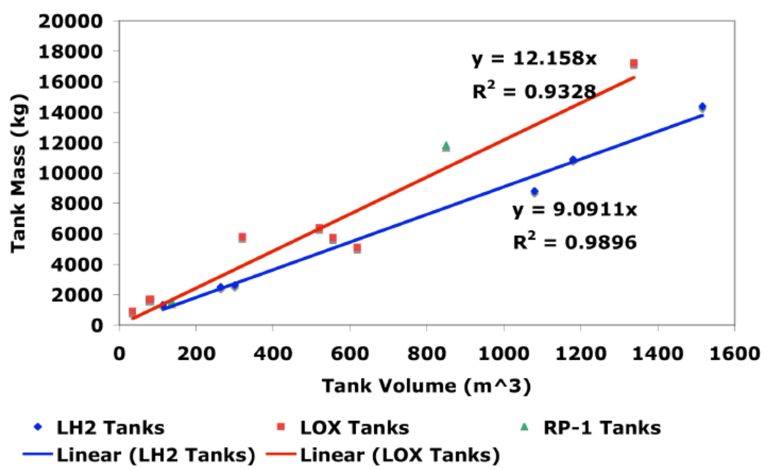


Figure 213: Propellant tank mass and volume correlation

As far as the propellant and oxidizer tanks are concerned, exploiting statistical data like those reported in Figure 213, proper tank sizing can be performed. In a first attempt, the overall volume can be estimated. Then, in order to comply with requirements of CG shifting, a series of trade-offs shall be carried out in order to find the best compromise between tanks dry mass and CG shifting.

6.4.2.2 Weight &Balance consideration

In this case, noticing the possible presence of propellant tanks, both dry and wet mass should be estimated. Moreover, besides the fact that this thesis focuses on a conceptual and preliminary design approach, the propellant depletion system shall be taken into account because it will provide useful information for the tank design. For this purpose, mission and flight simulations would provide very useful suggestions.

6.4.3 Tail cone Length estimation

The design of a tail cone is mainly driven by the fact that the configuration shall end somewhere guaranteeing the possibility of hosting special equipment, mainly the Emergency Locator Transmitter or some actuators to operate the rear-mounted Flight Control Surfaces. This will mainly impact the mass and CG evaluation while the Length estimation is primary driven by the need of closing the cone, starting from a certain diameter, i.e. the ending section of the system compartment or of the passengers' compartment (in case the system compartment is not available). The presence of a propulsion system integrated within the fuselage can also have a great impact on the sizing of this section. Indeed, the presence of one or more nozzles can impact on both the length and diameter.

As it is possible to be noticed in the flowchart reported in Fig. #, the length of the tail cone can be stretched in order to guarantee a proper arm to the control surfaces mounted on the tail. However, this design iteration can be done only in a following design step, once the wing and the fuselage have been integrated together and the aerodynamic centre of the overall configuration has been clearly identified and for this reason, this aspect will be further investigated in next chapter.

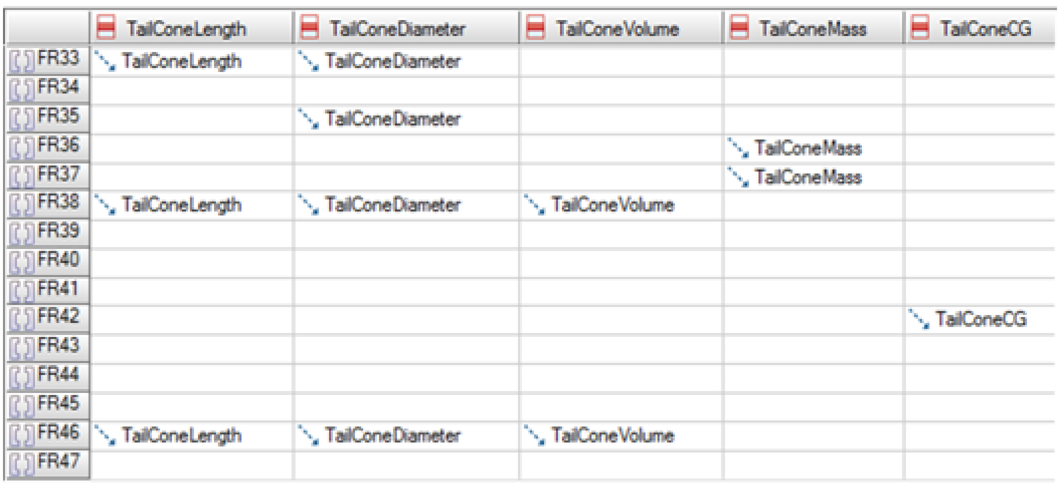
Different tail geometries may be considered, and their arrangement is strictly related to the controllability and manoeuvrability specific requirements.

Table 106: Requirements impact on fuselage tail section.

		Requirements impacting on fuselege	*Impacting Drivers*	*Impacted design parameters*	*Comments*
Comfort and Safety	*The fuselage shall safely accommodate crew*				
	The fuselage shall safely accommodate				
	The fuselage shall guarantee proper room allowing passengers to experience microgravity				
	The fuselage shall guarantee proper room allowing scientists to carry out experiments.				
	The fuselage shall guarantee a proper view of the				
Aerodynamic	*The fuselage shall generate the lowest possible drag.*		• Mach Number • Layout	D_TailCone L_TailCone	The drag produced by the fuselage tail cone can be minimized properly
	The fuselage heat loads shall be minimized				
	The fuselage wetted area shall be minimized.		Layout configuration	D_TailCone	shall be minimized in order to avoid a drag increase.
Structure	*The fuselage weight shall be minimized.*		• Material • Wet surface • TPS thickness and material • Primary structure thickness and	m_TailCone= f(m_struct, m_sys)	The mass of the fuselage tail cone can be evaluated considering its main geometrical features, the wall thickness and the asociated materials. In addition, the presence of subsystems to be integrated inside, such as an avionic bay with emergency equipment
	The fuselage shall sustain the structural loads all along the flight profile.		Structural loads profile	m_TailCone= f(t_wall)	The fuselage tail cone as well as all the different parts and components of the vehicle, shall sustain the loads all along the mission profile. To this purpose, proper material and thicknesses of the
	The fuselage shape shall be as symmetric as possible.		Layout configuration	Tail cone section shape	The fuselage tail cone section shall be as symmmetric as possible, meaning that the center of the section shall be as close as possible to vehicle longitudinal axis. This will not numerically impact on nose length or
	The fuselage shall accommodate propulsion				
	The fuselage shall accommodate avionic				
	The fuselage shall accommodate propellant				
	The internal arrangement shall guarantee the proper centre of gravity location.		• Integration of subsystems	CG_TailCone = f(m_structure)	The CG position of the fuselage tail cone is mainly dependent from the integration of the propulsion and
	The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.				
	The fuselage shall properly accommodate cargo				
Logistics and Operations	*The fuselage shall ease loading and unloading*				
	The fuselage shall guarantee proper airworthiness characteristics.		Regulations	Tail Cone section shape	The shape of the fuselage tail cone shall be properly hypothesized considering exiting airworthiness
	The fuselage shall be properly designed in order to allow visibility to the pilots.				

Table 107: Requirements allocation on the main fuselage passengers section design parameters.

	Tail Cone Length	Tail Cone Diameter	Tail Cone Volume	Tail Cone Mass	Tail Cone CG
The fuselage shall safely accommodate crew					
The fuselage shall safely accommodate passengers					
The fuselage shall guarantee proper room allowing passengers to experience microgravity					
The fuselage shall guarantee proper room allowing scientists to carry out experiments.					
The fuselage shall guarantee a proper view of the					
The fuselage shall generate the lowest possible					
The fuselage heat loads shall be minimized					
The fuselage wetted area shall be minimized.					
The fuselage weight shall be minimized.					
The fuselage shall sustain the structural loads all along the flight profile.					
The fuselage shape shall be as symmetric as					
The fuselage shall accommodate propulsion					
The fuselage shall accommodate avionic					
The fuselage shall accommodate propellant					
The internal arrangement shall guarantee the proper centre of gravity location.					
The fuselage shall allow the separation of a cabin escape system to prevent catastrophic events.					
The fuselage shall properly accommodate cargo					
The fuselage shall ease loading and unloading					
The fuselage shall guarantee proper airworthiness characteristics.					
The fuselage shall be properly designed in order to allow visibility to the pilots.					



	TailConeLength	TailConeDiameter	TailConeVolume	TailConeMass	TailConeCG
FR33	TailConeLength	TailConeDiameter			
FR34					
FR35		TailConeDiameter			
FR36				TailConeMass	
FR37				TailConeMass	
FR38	TailConeLength	TailConeDiameter	TailConeVolume		
FR39					
FR40					
FR41					
FR42					TailConeCG
FR43					
FR44					
FR45					
FR46	TailConeLength	TailConeDiameter	TailConeVolume		
FR47					

Figure 214: MBSE implementation of requirements allocation on the main fuselage tail section design parameters.

6.4.4 Length to Diameter ratio Optimization.

Considering the different flow charts presented for the different possible configuration alternatives, they have in common the presence of a block suggesting to carry out a Length to Diameter ration optimization. Indeed, both length and diameter are two of the most important parameters for the fuselage length. Please notice that this ratio, also known as slenderness ratio may be determined based on a certain number of specific requirements. In particular, referring to (Sadraey, 2012), the following major requirements with a possible impact on this design parameter can be elicited:

- The fuselage (L/D) parameter shall allow the lowest zero-lift drag.
- The fuselage (L/D) parameter shall guarantee the lowest wetted area.
- The fuselage (L/D) parameter shall minimize the fuselage weight.
- The fuselage (L/D) shall guarantee to maximize the fuselage internal volume.
- The fuselage (L/D) shall guarantee the lowest mass moment of inertia.
- The fuselage (L/D) shall enhance the aircraft stability.
- The fuselage (L/D) shall minimize production costs.

Carefully looking through this list it can be notice the high variety of Areas of Interest which these requirements belong. In particular, they range from aircraft performance to operations, controllability and manoeuvrability to end with a requirement strictly related to the manufacturing process. In addition, it has to be remembered that this list of requirements does not delete all the requirements used during the first sizing attempt, but rather they should be verified in addition. It is clear that depending on the specific mission, the relative importance of these requirements will vary. In any case, once the most important one has been selected, it becomes the optimization criterion and the designer is expected to develop a formulation to express the requirement in a mathematical form with length and diameter as two major variables. Then, differentiate the formula with respect to fuselage length or diameter. When the result of the differentiation is set equal to zero, the final solution yields the optimum fuselage length to diameter ratio.

For example, it is possible to demonstrate that with the objective of minimizing the zero-lift drag, an optimum slenderness ratio $(L/D)_{opt} = 16.3$ has been obtained (Sadraey, 2012). Complementary, the optimum fuselage slenderness ratio to minimize the fuselage surface area is just 1. Therefore, it is desirable to have a short fuselage to minimize the fuselage surface area, while it is desirable to have a long fuselage to minimize the fuselage zero-lift drag.

Eventually, it is clear that the selection of the optimal value of slenderness ratio shall be properly traded-off depending on the specific set of mission requirements.

6.5 Application to the case-study

This section aims at reporting the application of the methodology presented in this chapter to the case study dealt with in this Thesis. Thus, following the activity flow suggested above, this section is articulated into several subsections, starting from the identification of the fuselage layout, moving to more quantitative evaluations, up to focusing on the detailed definition of the Cabin Interiors' layout.

6.5.1 Fuselage Configuration Selection.

Once the driving requirements are known as well as the mission the vehicle shall carry out, a proper Fuselage Configuration shall be selected. In this case, after proper evaluations, the Configuration 5, hosting:

- Crew Compartment
- Passenger Compartment
- Rocket Engine
- Propellant Tanks

has been selected.

This is the starting point for the following, more quantitative evaluations that allows defining the overall external shape of the vehicle.

6.5.2 Nose and cockpit sizing.

In order to start with the design of the nose, the following input data and hypotheses has been considered:

- $k_{nose} = 0{,}8$ hypothesizing a blunt nose with a medium-high radius of curvature;
- $k_{nose\ add} = 0$ hypothesizing no additional empty space.

- $R_1 = 0{,}4\ m$ is the radius of the blunt edge.
- $D_{cockpit} = 2{,}8\ m$ in accordance with some statistical vehicles considered as reference.
- $M = 2$ on the basis of the results of the high level mission simulations.

It has to be noticed that no assumptions have been made as far as the $\alpha_{overnose}$ is concerned. This is only due to the fact that the suggested formulation cannot be applied for this specific case study because it is a VTOL aircraft.

Starting from these first set of assumptions, the following output values has been obtained: $L_{nose} = 3{,}07\ m$ and $D_{nose} = 2{,}8\ m$. However, after a first estimation, it has been noticed that the obtained nose length is sufficient also to host the cockpit section, considering that in the nose section, no peculiar sensors have been envisaged. For this reason, a standard cockpit of about 1m length and a maximum diameter of 2,8 m has been included in the volume evaluated for the nose section, as it is displayed in Figure 215, allowing the comparison between a previous and a second iteration.

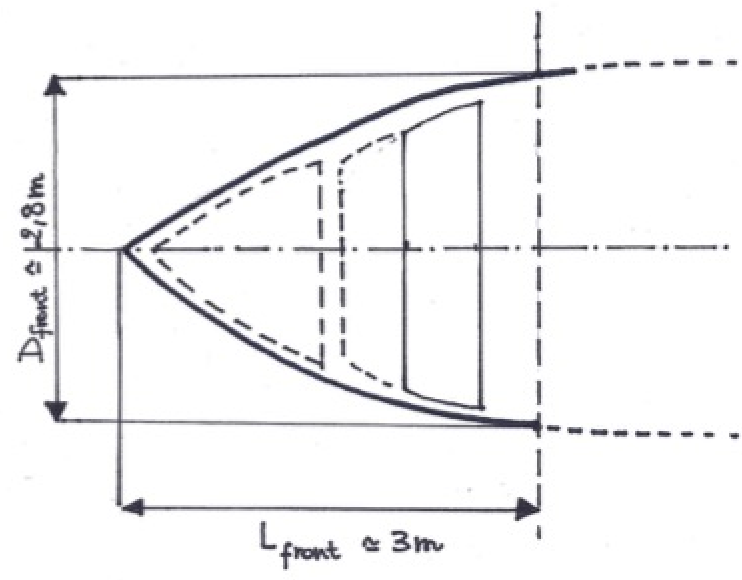


Figure 215: Front Fuselage sizing for the reference case study

These high-level evaluations can be validated in different ways. In particular, in this case, CAD and X-Plane flight simulator can be used in order to validate.

6.5.3 Crew Compartment sizing.

As far as the crew compartment sizing is concerned, proper level of comfort shall be guaranteed to the pilots. For this reason $L_{crew\ comp}$ of 1 m has been assumed. Complementary, considering that in this case, the avionic bay can be easily integrated within the available volume, a 2,8 m diameter has been maintained.

6.5.4 Passenger Compartment sizing.

The passengers' compartment size is mainly affected by the number of passengers and by the related level of comfort that should be guaranteed. In this case, the sizing algorithm presented in Section # can be applied assuming that:

- $n_{pax} = 2; n_{decks} = 1; n_{classes} = 1.$
- $n_{aisle} = 1; w_{aisle} = 0{,}7\ m; w_{seat} = 0{,}7\ m; h_{aisle} = 2{,}10$
- The passengers compartment should be designed considering that enough free volume should be guaranteed allowing passengers to experience microgravity and free-floating within the available space. Moreover, additional volume should be considered thinking that, at least at the beginning, the passengers will wear proper suits, with integrated helmets and for this reason, their mobility will result slightly reduced. A small wardrobe will be implemented.
- The mission is so short that only a small toilet will be designed. However, galleys are supposed to be substituted by ad-hoc single meals given to each flight participants.

Taking a look at the available regulations in civil transportation, in case of a so small group of passengers, the main door may be sufficient. However, a small emergency exit door has been designed and allocated in front of the access door, 1 m wide, allowing an optimal exploitation of the volume, avoiding an excessive stretching of the fuselage. In addition, a small toilet has been included, hypothesizing a length of 0,7 m.

As far as the passengers' seats sizing is concerned, the need of hosting special support equipment for the passengers shall be taken into account, at list during the first years of operations, when these vehicles and related missions will be considered experimental. For this reason, additional design margins have been used:

- For the access door: to the minimum length of 0,6 m additional 0,4 m have been included to allow passengers to facilitate boarding and un-boarding operations of passengers with space suite.

- For the seats, different trade-off activities have been carried out considering the special needs of allowing passengers to float during the period of microgravity. For this reason, standard dimension seats have been selected but additional free volume of 1,3 m has been considered. As it is demonstrated by the in-depth evaluations reported in the next section, the volume available to experience microgravity has been verify with CAD models as well as through simulations.

Following the algorithm proposed in the previous sections, the following outputs have been obtained (Figure 216):

$$L_{PaxComp} = 3{,}5\ m$$

$$D_{PaxComp} = 2{,}8\ m$$

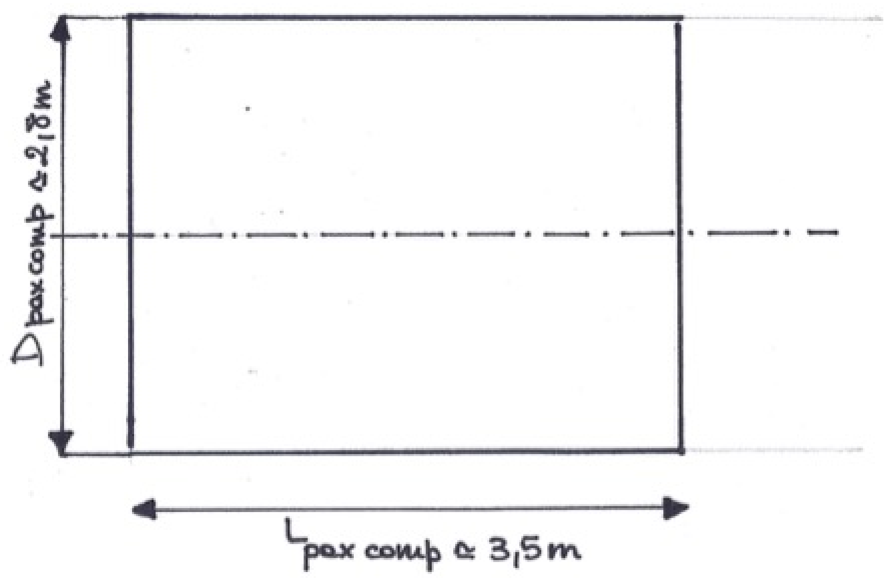


Figure 216: Passengers compartment sizing for the reference case study

6.5.5 Aft Fuselage sizing.

The aft fuselage of this case study shall be able to accommodate a rocket engine and relative propellant and oxidizer tanks. In this case, the relative location of tanks and rocket cannot be negotiated, considering that the rocket nozzles shall be placed at the end of the fuselage and in order to avoid increasing in section dimensions, the tanks will be hosted in a proper compartment standing between the passenger’s compartment and the rocket engine.

As far as the rocket sizing is concerned, considering the level of thrust and specific impulse required and its mode of operations, an existing reference engine has been selected and its major dimensions considered for the aft fuselage sizing. Complementary, the exploitation of mission simulations allows the designer to know the total amount of propellant to be stored on board. In addition, knowing the mass and the specific densities, it was possible to define the volumes for the propellant and oxidizer.

Considering that the selected configuration consists of a rocket motor surrounded by two air-breathing engines with propellant and oxidizer tanks hosted between passengers’ compartment and the rocket, the layout of Figure # has been obtained. In the next tables, the main data related to the propulsion and propellant subsystems are summarized. Then, starting from these main data, the overall aft fuselage sizing can be carried out.

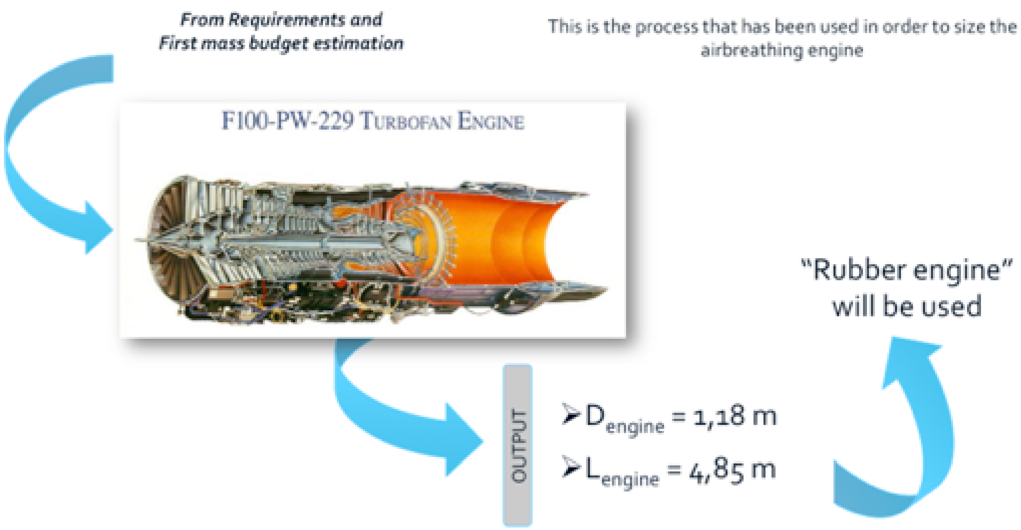


Figure 217: Air-breathing engine sizing for the reference case study

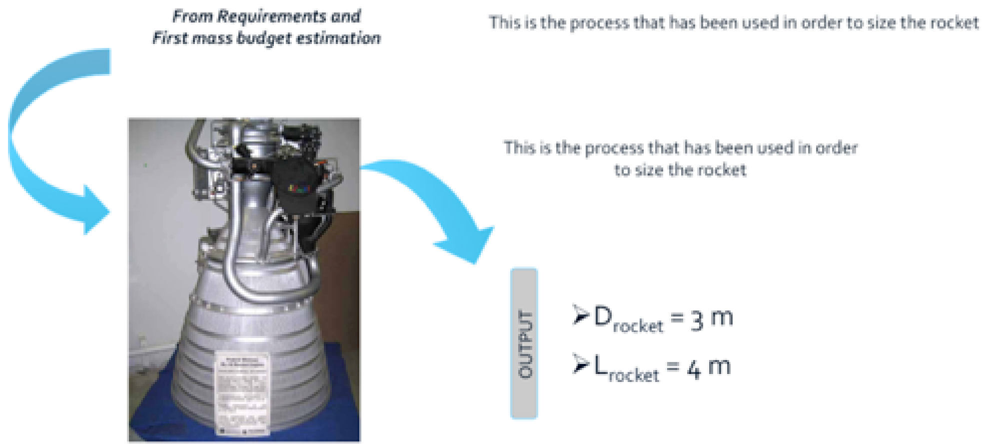


Figure 218: Rocket motor sizing for the reference case study

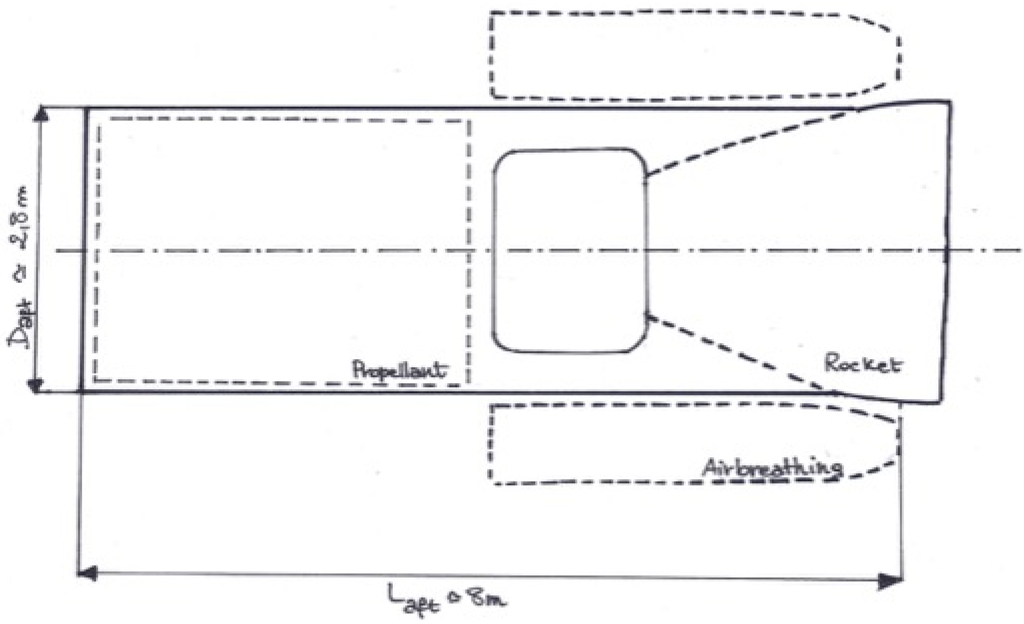


Figure 219: Aft Fuselage sizing for the reference case study

Table 108: Summary of propulsion system sizing results for the reference case study

Characteristics	***Unit (SI)***	***Value***
rho_LOX	kg/m3	1140
rho_LH2	kg/m3	71
m_TOT	kg	5200
m_LOX	kg	4457
m_LH2	kg	743
V_LOX	m3	3,9
V_LH2	m3	10,5
d_tank_LOX	m	0,8
L_tank_LOX	m	3,5
# tank LOX	-	2
d_tank_LH2	m	0,8
L_tank_LH2	m	3,5
# tanks LH2	-	6

Eventually, the following results have been obtained:

$$L_{aft} = 8\,m$$

$$D_{aft} = 2{,}8\,m$$

Please, notice that in this case, the diameter refers to the fuselage main structure. However, the rocket nozzle will exceed in length, allowing a proper integration of the nozzle. In addition, as it is sketched in Figure 219 the air-breathing engines are located closed to the fuselage-wing connection and proper intakes should be designed closed to the conjunction of the wing leading edges and the fuselage structure. Additional details are provided in the next section, specifically dealing with the wing-fuselage integration.

In the next two pictures, an overview of the complete fuselage layout is reported.

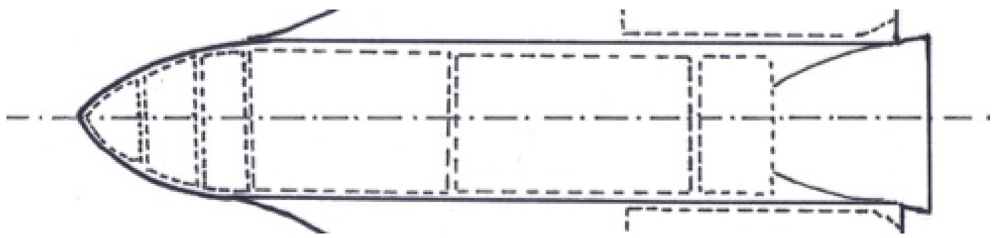


Figure 220: Fuselage layout for the reference case study

6.5.6 Cabin Internal Layout

This section is the result of a collaboration between Thales Alenia Space – Turin and Politecnico di Torino in which the author was active participant.

The requirements and guidelines used for the internal Cabin envelope definition are summarized hereafter:

- External cabin dimension
- Emergency hatch in addition to the entry door
- Cabin acting as escape capsule in case of emergency during Microgravity mission
- Subsystem required volumes
- Safety (i.e. space suit interface for Microgravity experience)
- Pilot needed volume
- Passengers number
- Maximize passengers comfort for the for both microgravity experience and Point to Point mission
- Facilities to be provided

As a first step, starting from the cabin envelope, defined in the previous section, the emergency hatch positions and dimensions in addition to the entry door (acting also as emergency door) has been identified with help of a CAD model. The introduction of an additional opening reduces the available volume for the subsystems, and it has a deep impact on the cabin structure itself. For this reason, it has been decided to introduce a hatch instead of an additional emergency door, in respect of aeronautical regulations but also looking towards more space-related architectures. The hatch will allow passenger escape in case of emergency and at the same time is reducing the required opening on the cabin structure. The hatch position has been defined to guarantee passengers egress from the cabin taking also into account the possibility of a splash down in case of contingency (Figure 221). As a matter of fact, the passenger compartment is located above the intermediate part of the cabin and in case a CES will be implemented, in case of splash down, the overall system should remain above the floatation line. From a more theoretical perspective, this is a perfect example of high level requirements (safety) impact on specific component level design parameters (emergency door location).

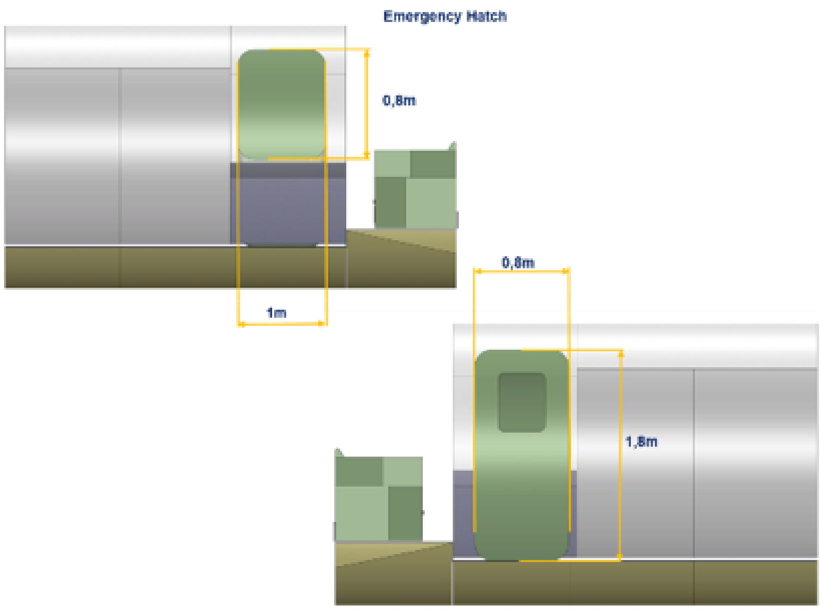


Figure 221: Entry and emergency door location and hatch

From Figure 221 it is also possible to notice that the entry door is including the unique window on the vehicle (with the exception of the cockpit windows). The choice to limit the number of windows is mainly driven by structural constraints due to the expected demanding aerothermodynamic loads.

As far as structure design is concerned, the airframe and shall be primary designed to guarantee that the pressure environment inside the vehicle is maintained. In order to avoid any pressure leakages, in particular in the areas where the hatches are planned to be installed, it will be mandatory to reinforce/stiff the vehicle’s structure to limit the structure’s deformations. In fact, high structure's deflections may induce high pressure’s leakages inside the vehicle.

The design complexity of Hatch’s Systems (i.e. mechanisms to allow opening/closing of the doors), as well as windows, may also require a reinforcement of vehicle’s structure to sustain the loads/stresses induced during the flight phases. Unfortunately, these reinforcements will increase the vehicle dry mass, reducing the mass available to host payload, passengers or to install subsystems. In addition, in a snow-ball effect, the increase in the vehicle dry mass will imply a higher fuel consumption.

A good design compromise to satisfy both the stakeholder expectations of looking outside and the mass constraints would be the adoption of O-LED panels. Please, notice that this design solution will maximize microgravity experience by creating a 360 degree space landscape (EVA like).

As a second step in the cabin internal dimension definition process, passengers number and cockpit position have been considered. It has been decided to have passengers and pilot cockpit on different levels. The main reasons are summarized hereafter:

- Assure a better view to pilot and co-pilot during flight, landing and take off
- Increase the available volume underneath the floor to be used by subsystems
- Assure a passenger area cabin height of about 2000mm allowing passengers to stand up

The guidelines used to define passenger needed volume are listed here below:

- Seat dimension and possibility to recline them during point to point mission (passenger comfort),
- Nominal passenger number taking into consideration the possibility to increase it for point to point mission,
- Possibility to include facilities (i.e. toilette and luggage compartment),

- Minimum volume needed during microgravity experience

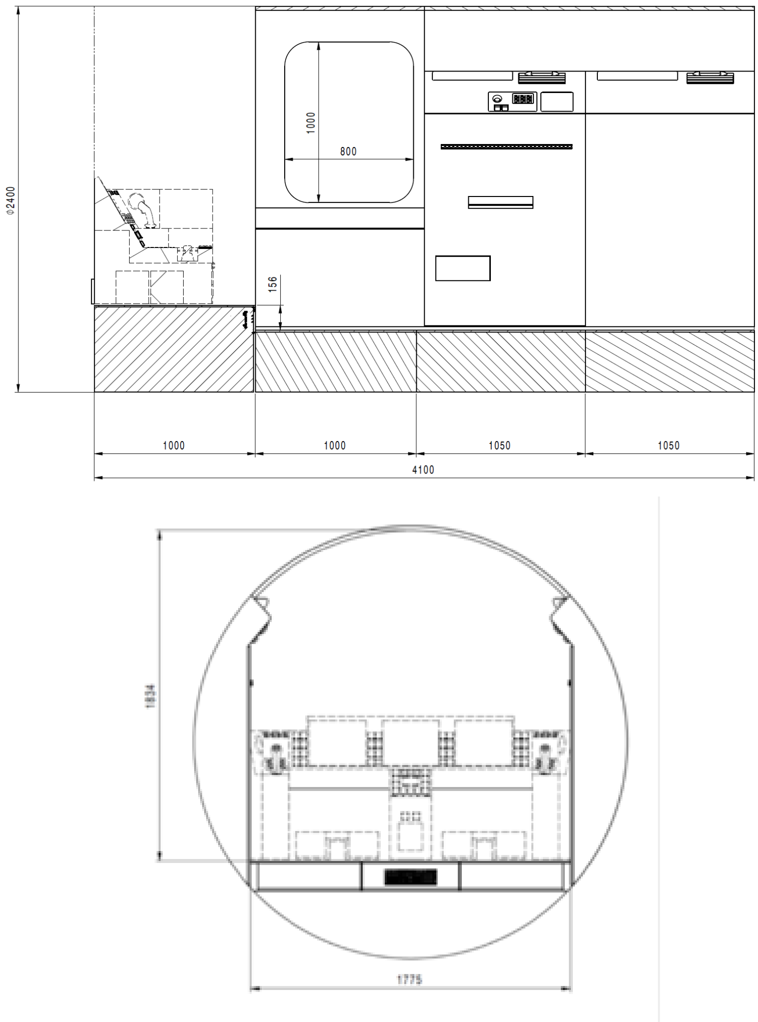


Figure 222: Lateral and section View of the Crew and Passengers compartments after CAD modelling

The remaining volume, available for subsystem, can be derived deducting the just defined internal envelope from the external cabin envelope. Based on the evaluation this volume should be of about 3,7m^3 (Figure 223).

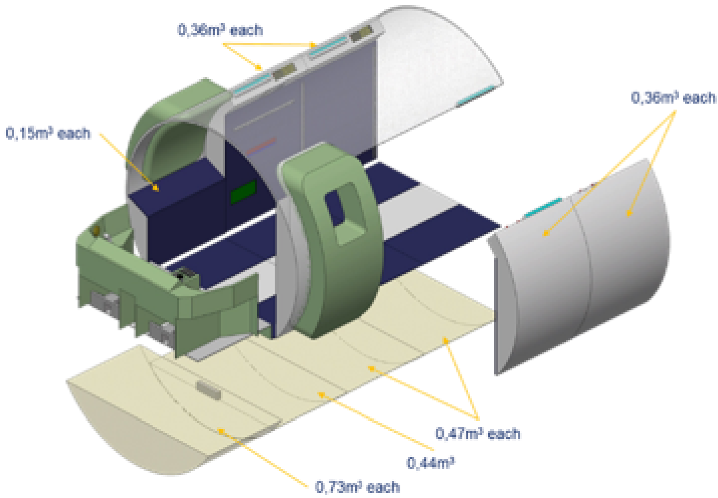


Figure 223: Available volume for subsystems integration

Different trade-offs have also been performed to identify the final layout to be used during the two missions. The chosen configurations are shown in (Figure 224, Figure 225). As listed above, the internal cabin design took into account the possibility to quickly turn one configuration into another. For this reason, the possibility to have removable panels, that could be disassembled when not needed for the chosen configuration and replaced with the correct one has been conceived. In some cases, instead of replacing the entire panel, the possibility of covering specific area with dedicated panels have been envisaged too. For example, the toilette walls can be removed to switch from the point to point configuration to the microgravity one, the remaining interfaces will be covered with a dedicated panel. The same concept has been considered for the suit control panels that shall be available during microgravity mission but they will be covered in the point to point configuration. The proposed internal layout is also providing utilities common to both missions. As shown in the following picture, the proposed internal layout identifies lights and air intake preliminary positions. In particular, one the air intake is taking advantage of the available volume between passenger and cockpit floors, the other four are placed on the top side of the internal volume to allow a proper air flow.

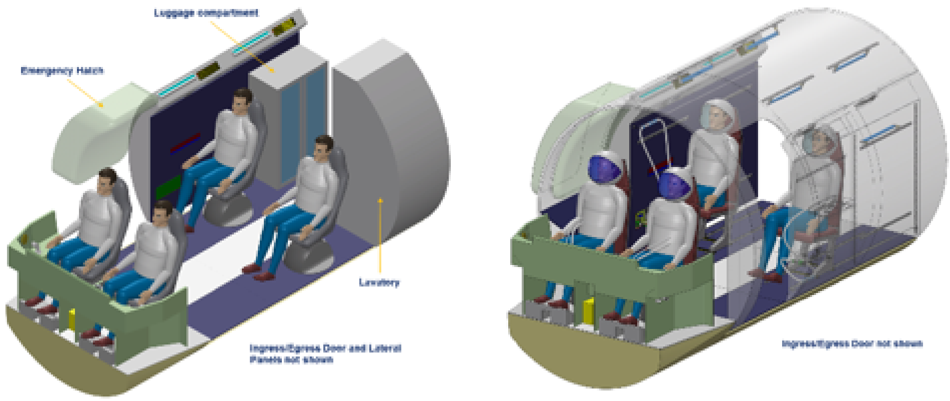


Figure 224: Point to Point Mission Cabin Layout vs Microgravity Mission Cabin Layout - Flight Configuration

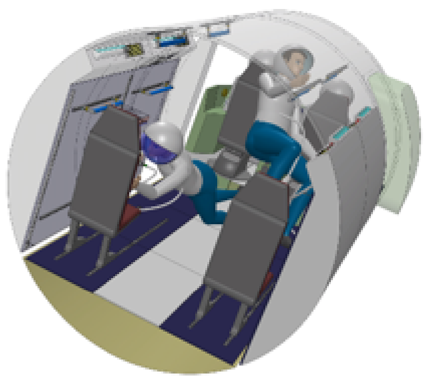


Figure 225: Microgravity Mission Cabin Layout – Microgravity Experience Configuration

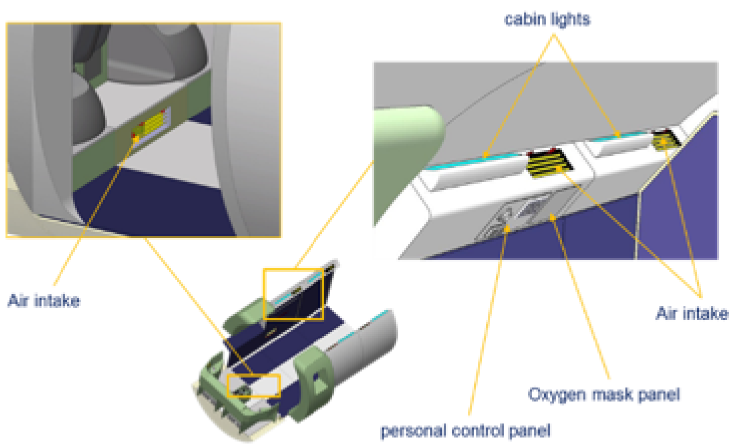


Figure 226: Constructional details

For each panel the internal layout is also providing a personal control panel with a light and an air flow inlet and an oxygen mask panel (Figure 226). The oxygen mask panel is available, during point to point mission while for the microgravity mission is not engaged since passengers wear space suits and helmets. During microgravity experience passengers could have a limited mobility inside the cabin for this reason crew restraints have been considered: handrails and foot-restraints will be properly located inside the cabin. Those interface will be removed in case of point to point cabin configuration. Microgravity mission may require to simultaneously have passengers and scientific payload installed, for this reason seats positions takes into account the occupied scientific payload volume. Few configurations have been studied and the one with the minimum impact to the volume devoted to the passengers during microgravity allows seats movement is reduced of about 200mm and allows to accommodate the scientific payload too. The nominal seat displacement may be restored with scientific payload installed as shown in the following figure. At the present time it is not clear whether the proposed payload configuration is feasible or it is not.

The preferred configuration valid for 4 persons (2+2) represents the best compromise between number of passenger and cabin given dimensions. It maximizes the distance between passengers and cockpit area and allows the accommodation of a lavatory and a luggage compartment in the rear part of the vehicle.

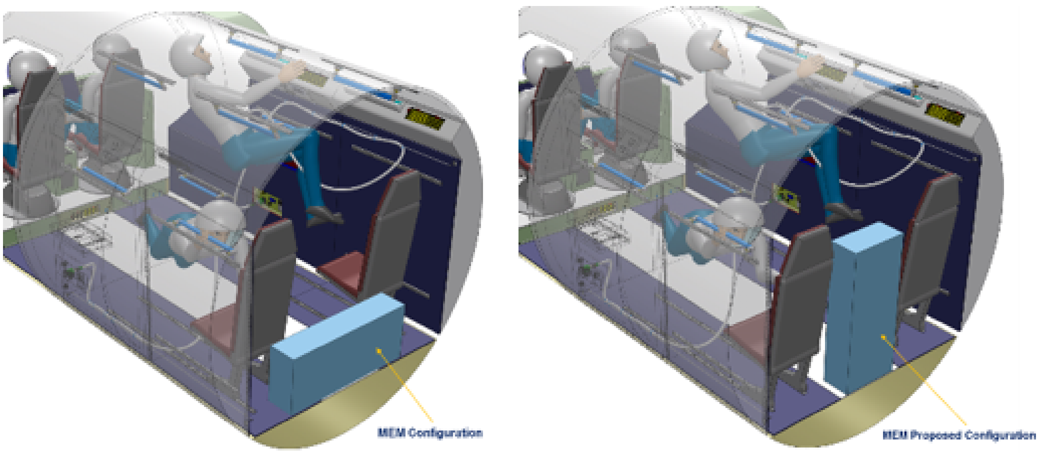


Figure 227: Different scientific payload location

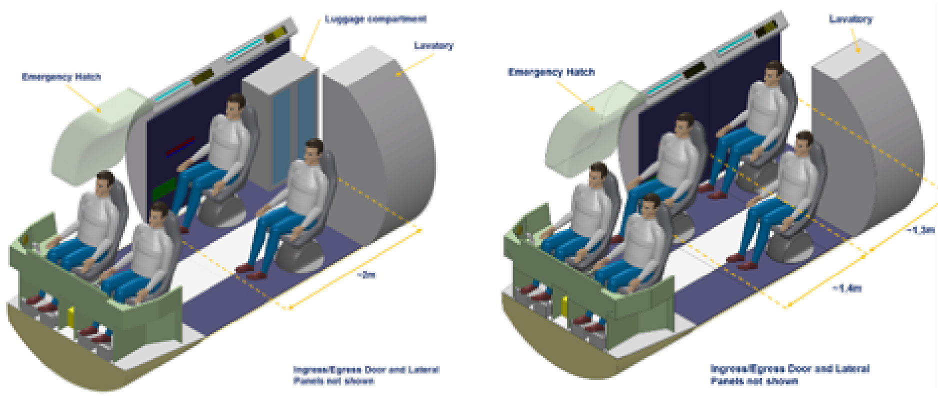


Figure 228: 2+2 conf. vs 2+3 conf.

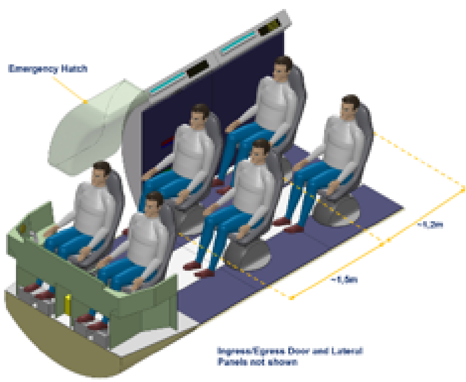


Figure 229: Extreme passengers configuration

During the CAD model development, special attention has been devoted to the seats for passengers to allow safe accommodation during both the pure transportation phases and the microgravity experience. In particular, following a proper SE methodology the following list of requirements should be considered

1. The seat shall be able to rotate on their axis in order to move their back on the cabin lateral wall during microgravity experience
2. The seat shall be equipped with folding back
3. The seat shall be equipped with a proper mechanism able to pull down the entire seat during the microgravity experience

4. The seat shall be mounted on ad-hoc rails to allow passengers to move them toward to the rear part of the cabin during microgravity experience

Also in this case, different options have been envisaged. In particular, the very first (Figure 230) suggested configuration assure the availability of seats at any time during the microgravity experience and it may not require to be manually operated (in case the rotation is driven by a mechanism). This design allows to have seats immediately available in case of any kinds of passenger discomfort. Moreover, the seat interface is in common to the one foreseen for the point to point mission. On the other hand, the internal cabin volume results divided and therefore this layout is not representing the best choice in terms of volume exploitation.

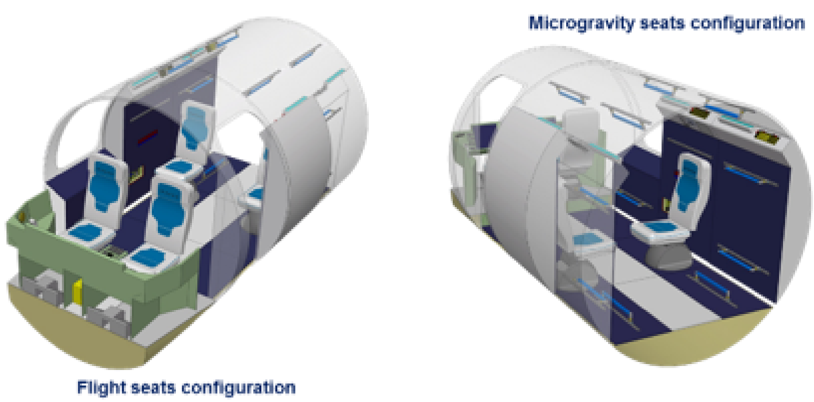


Figure 230: Microgravity seats Conf. 1

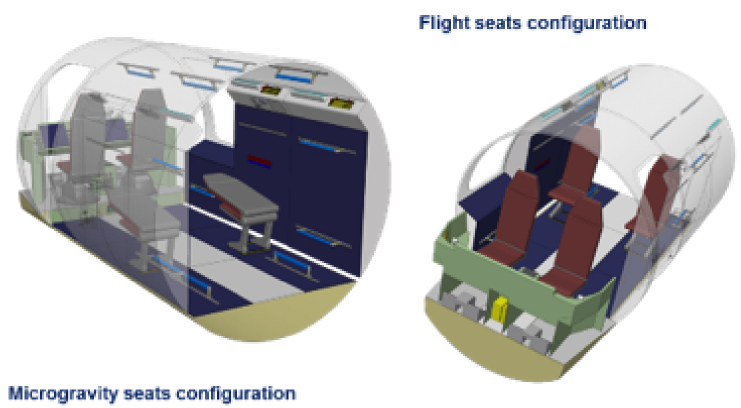


Figure 231: Microgravity seats Conf. 2

The second seats configuration (Figure 231), as well as all the other ones that have been suggested, are mainly based on lightweight helicopter design. The shape is very simple and therefore the occupied volume is minimum. Conf. 2, provides seats with manual folding back, increasing the available volume for microgravity, without optimizing it. In addition, seats will not immediately available in case of passenger sickness.

The third configuration can be considered as an evolution of Conf. 2 seats layout. In this case the seat has a folding back, but it also includes a mechanism able to pull down the entire seat to reduce the total height during the microgravity period. This option increases the internal available volume, but it increases the complexity of the seat design since it requires an ad-hoc mechanism able to lower down the seat and at the same time strength enough to support the landing impacts in case an emergency.

The last suggested alternative (Figure 232) envisages seats installed on dedicated rails and they can be manually moved on the rear part of the vehicle. An automatic device has been excluded preventing volume reduction underneath the floor. This kind of configuration maximizes the volume available during the microgravity and, even if it requires to be manually operated to restore the flight position, it assures the seat availability at any time. In addition, this configuration is also compatible with the option requiring scientific payload installed during passenger microgravity with a minimum volume impact.

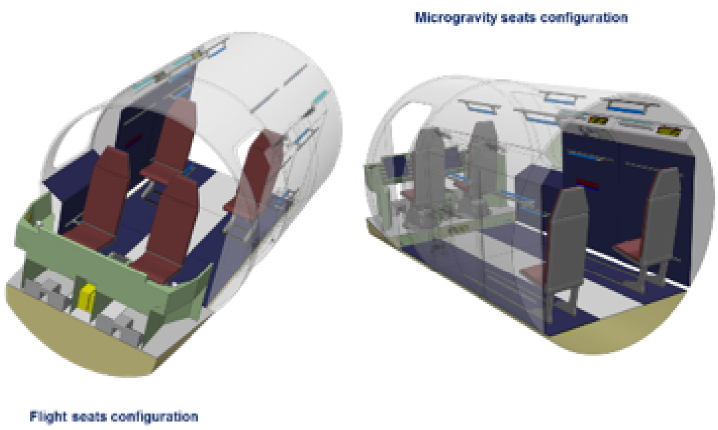


Figure 232: Microgravity seats Conf. 4

Based on all these considerations, a trade-off has been carried out following the same process already depicted in previous sections, the last alterative resulted to be the best option to accommodate passengers during suborbital parabolic flight.

Furthermore, a proper lavatory shall be envisaged. Its design is based on a rack confined in the rear part of the cabin. The design approach allows to remove/install the rack wall, doors and equipment keeping the subsystem interfaces fixed on the cabin. Those interfaces will be covered by a dedicated panel in the microgravity cabin configuration. The proposed design will simplify the activities required to turn the microgravity configuration into point to point or vice-versa. In front of the lavatory, a small luggage compartment may also be provided.

Figure 233: Lavatory design (left) and integration (right)

Eventually, considering that microgravity missions may require suits for passengers and pilot, the interior design of the cabin provides dedicated interface panels to connect suit umbilical to cabin life support (Figure 234 and Figure 235).

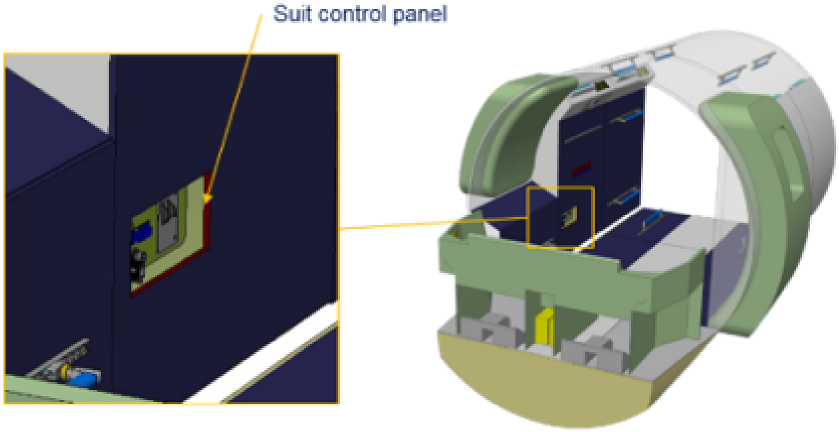


Figure 234: Suit Control panel detail (left) and integration (right)

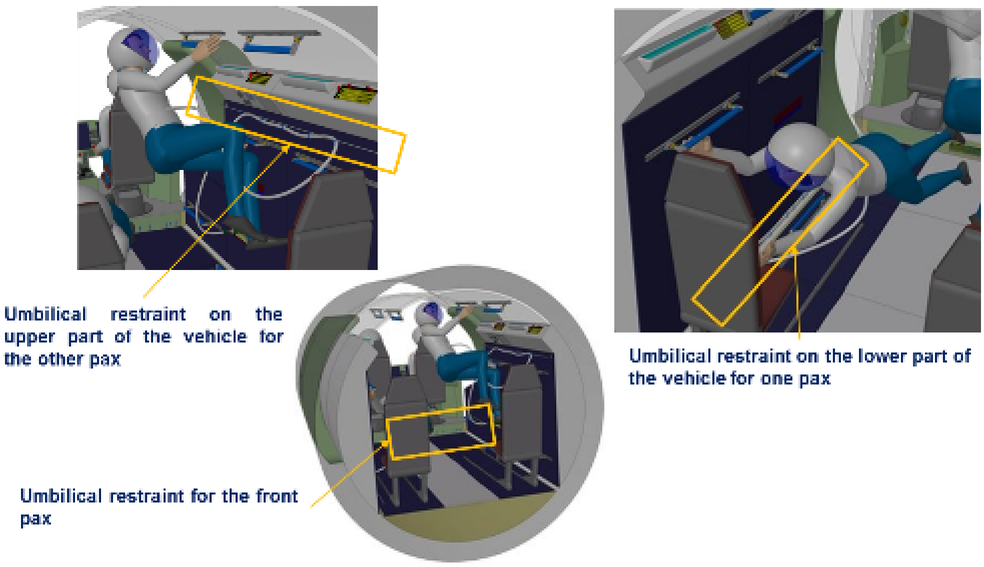


Figure 235: Umbilical restraint

References

Bunton, D. «Generic moves in PhD thesis introductions.,» in Academic discourse, London, Pearson Education Limited, 2002, pp. 57-75.

EASA (2017), Certification Specification for Large Airplanes

Fusaro, R., et al. (2015). Hyperspectral sensor for civil applications to Unmanned Aerial Vehicles within a monitoring scenario. *2015 IEEE 1st International Forum on Research and Technologies for Society and Industry, RTSI 2015* - Proceedings, art. no. 7325146, pp. 490-497.

Fusaro, R., et al. (2016) Selection of an uav configuration layout for optimisation of teledetection mission using hyperspectral sensors (2016) *30th Congress* of the International Council of the Aeronautical Sciences, ICAS 2016 (Daejeon, South Korea)

Kwan, B. S. «Reading in preparation for writing a PhD thesis: Case studies of experiences,» Journal of English for Academic Purposes, pp. pages 180-191, 2009.

Raymer, D. P., (2012). Aircraft design: A conceptual approach, *American Institute of Aeronautics and Astronautics*. Inc., Reston, VA (2012): 21.

Roskam, J. (1985). Airplane design. *DARcorporation*,.

Sadraey, Mohammad H. (2012) Aircraft design: A systems engineering approach. *John Wiley & Sons,* 2012.

Steelant, J., et al. (2015). Achievements obtained for sustained hypersonic flight within the LAPCAT-II project. *20th AIAA International Space Planes and Hypersonic Systems and Technologies Conference. 2015*.Glasgow (UK)

Chapter 7

Major integration design issues

7.1 Wing and Fuselage integration major challenges

This chapter aims at suggesting useful approach in order to complete a first design iteration, facing the noticeable problem of integration, starting from the selection of the optimal mutual longitudinal position of wing and fuselage, moving to additional related challenges, such as the integration of other fundamental elements such as the landing gear and the empennages and the optimal sizing of control surfaces. Figure 236, summarizes the major challenges in terms of integration to be faced with at the beginning of the design process, suggesting a useful approach to be followed. This chapter tries to cover all these steps focusing on the major activities to be carried out.

In particular, integrating one or more elements of an aerospace products means aiming at optimizing the structural interfaces with the goal of minimizing the overall mass, maximizing structural and aerodynamic performances. For this reason, this is an example of multidisciplinary activity with a very high level of complexity that should be properly tackled with an organized and structured approach, like the Systems Engineering. For this reason, following the same approach already applied all along this Thesis, the integration activity is carried out in a MBSE environment, and led by requirements elicitation, verification and update process.

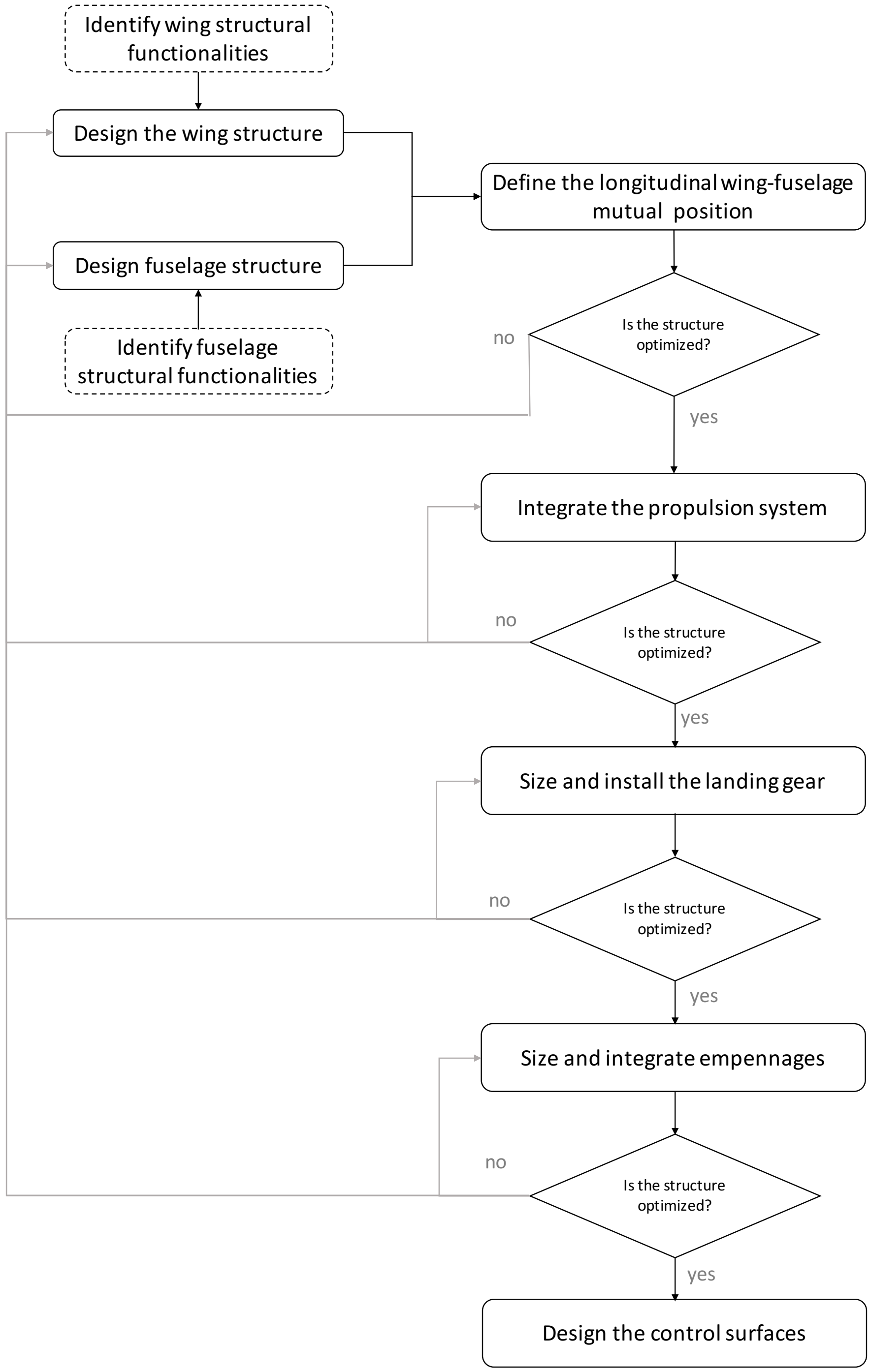


Figure 236: Flow-chart for the integration of the main aircraft components in conceptual design

7.2 Wing and Fuselage structural integration

Once the wing and the fuselage have been sketched and sized, even if at conceptual design level, it is important to integrate them to start having an idea of the overall vehicle. As it is shown in Figure 236, the first activity to be performed is the definition of the structure of both wing and fuselage. Then, once the structure has been defined, depending on the integration strategy already selected in Chapter 5, e.g. carrythrough wing box or not, etc., proper structural solutions should be envisaged.

In general, from the structural point of view, there are five major stresses to which all aircraft are subjected:

- Tension, that is the stress that resists a force that tends to pull something apart. For example, the engine tends to push the aircraft forward but the air resistance counteracts holding it back.
- Compression, the stress that resists a crushing and tends to shorten or squeeze aircraft parts.
- Torsion is the stress resulting in twisting. For example, while moving the aircraft forward, the engine also tends to twist it to one side, but other aircraft components hold it of course, creating torsion.
- Shear is the stress that resists the force tending to cause one layer of a material to slide over an adjacent layer. For instance, two riveted plates in tension subject the rivets to a shearing force.
- Bending stress is a combination of compression and tension.

In general, each single member of the structure may be subjected to a combination of different stresses. However, in most cases, the structural components are designed in order to carry end-loads rather than side ones, thus, they are designed to be subjected to tension or compression rather than bending.

Thus, starting from these considerations, it is clear that the definition of the fuselage and wing structure is mainly affected by two Areas of Interest:

- Structure and airworthiness
- Weight and balance.

Indeed, the major functions to be carried out by the fuselage structure can be summarized as follows:

- To support structural loads during the overall mission profile
- To support concentrated loads due to elements attachments
- To maintain the fuselage shape
- To support tension stress
- To support compression stress
- To support torsion stress
- To support shear stress
- To support bending stress

It is clear that it is not possible to allocate each function to a single element and this is mainly due to the fact that it is the integration of all these structural elements that allows supporting structural loads. However, this analysis can e considered the starting point to define the fuselage internal structure.

It is clear that the size of each element is strictly related to the interaction of loads insisting on it and proper structural models should be implemented but these data is not always available at conceptual design stage. It is mainly for this reason that in the next two subsections, practical guidelines are suggested allowing the definition of the fuselage and wing structure from a more qualitative perspective, giving some numerical suggestions coming from literature for the sizing of the major elements.

7.2.1 Wing structure

Following a Systems Engineering approach, in order to define the wing internal structure, the major functionalities may be derived allowing the elicitation of the main structural elements required to perform these functions.

Indeed, the starting point should be the identification of the primary functions of the structural elements of a wing, taking also a look at the various configurations that can be used. Often wings are fully cantilevered, meaning that no external bracing is need and the internal wing structure, with the help of the skin of the aircraft, shall be able to entirely support the loads. However, there are also other wing configurations that exploits external struts and wires to assist in supporting the wing and carrying the aerodynamic and landing loads but these designs are not investigated in the detail in this context because they are not compatible with the basic aerodynamic requirements for hypersonic vehicles.

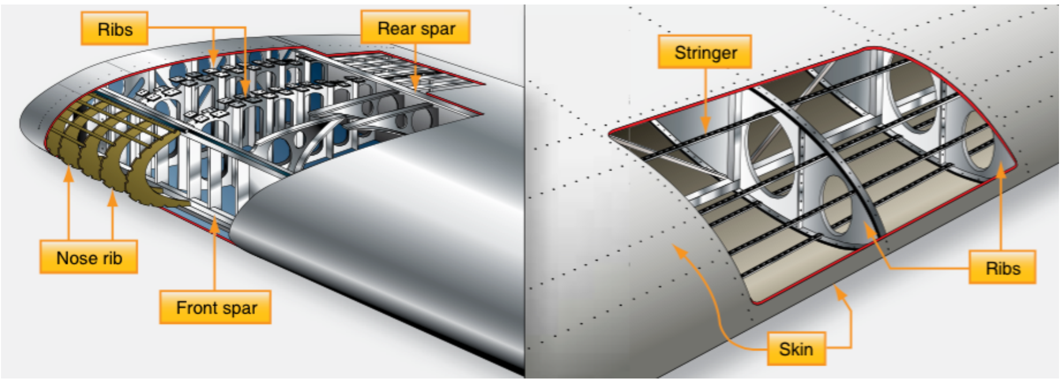


Figure 237: Main wing structure members

From the constructional perspective: wing can have three different designs:

- Monospar
- Multispar
- Box Beam

The monospar wing incorporates only one main spanwise or longitudinal member and its construction. Ribs or bulkheads allows the wing to maintain the airfoil shape in each wing section. This design is typically modified adding false spars or light shear webs all along the trailing edge with the aim of supporting control surfaces for example.

The multispar configuration incorporates more than one main longitudinal structural element and maintain ribs and bulkheads to guarantee the airfoil shape through the different wing stations.

A different type of design is the box beam. It exploits two main longitudinal structural elements with connecting bulkheads to guarantee additional strength, besides the normal function of giving contour to the wing. Corrugated sheet may also be placed between the bulkheads and the smooth outer skin layer so that the wing can better carry tension and compression loads. In some special cases, this strategy can also allow to substitute heavy longitudinal stiffeners.

In a box beam arrangement, spars play a crucial structural role, such as longerons in fuselage because when other elements of the wing are placed under load, most of the resulting stress is passed on the wing spar. They usually run parallel to the lateral axis of the aircraft from the fuselage toward the tip of the wing and are usually attached to the fuselage by the wing fittings, plain beams or truss. From a constructional point of view, different shapes can be envisaged: solid, box shaped, partly hollow or I-beam. As a rule of thumb, it can be considered that each wing has two different spars: one usually located in the front part of the wing and the other one at two third of the wing chord. Another important role is played by ribs that combine with spars and stringers to make up the entire framework of the wing. They are responsible of transmitting loads from the skin and stringers to the spars. In addition, depending on the specific location, each rib can have a proper specific functionality. For example, those placed entirely forward the front spar are used to shape and strengthen the leading edge. The skin part of the wing is designed to carry part of the flight and ground loads but only in combination with ribs and spars.

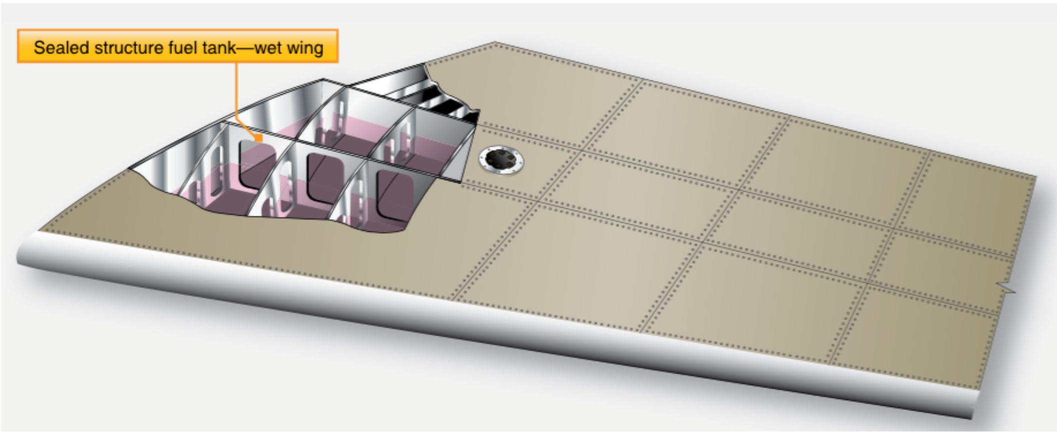


Figure 238: Internal wing detail: integrated fuel tanks

Furthermore, as it has been underlined in the chapter dealing with wing design (Chapter 5), the wing usually hosts fuel and to this purpose special design may be envisaged. Indeed, the joints in the wing can be sealed with special sealants enabling the fuel to be stored directly in contact with the primary structure. Besides this is the design that represents the optimum, from both the structural and weight points of view, alternatives exist. For example, a fuel-carrying tank can be properly fitted inside the wing volume (Figure 238).

This analysis facilitates the elicitation of the functions and of the list of requirements to lead the design and sizing process.

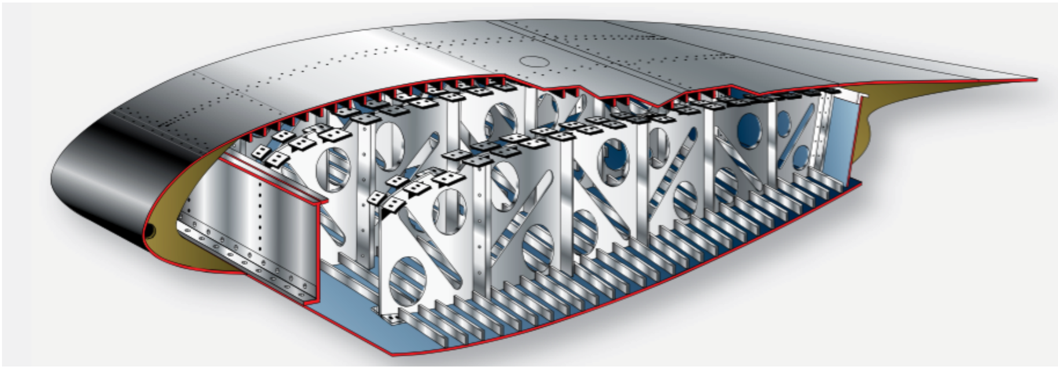


Figure 239: Example of complex wing structure

In order to support these activities, in addition to the functional model, in any case essential, as well as the requirements database, a physical model, exploiting CAD model should be generated. The creation of a CAD model will result to be essential for several follow-on activities related to physical, functional, behavioural and dynamic simulations.

In order to sketch a wing internal structure, the following activity flow is suggested. In particular, considering that for hypersonic vehicles, the Box Beam strategy seems to be the most promising, the following steps hide the selection of the box beam constructional configuration. However, similar procedures may be adopted to sketch the other configurations.

1. Define the wing geometry and external layout.
2. Sketch the location of the front and rear spar working in parallel with top-view and section-view (the intersection with the fuselage can be adopted). The front spar location may be located in correspondence to the airfoil station with the major thickness. In case leading edge mobile surfaces are planned to be installed, the front spar may be located in such a way that it can support it. The rear spar location can be put at 2/3 of the chord, allowing the integration of trailing edge mobile surfaces.
3. Ribs shall be located at a regular distance, that may depend on the different loads to be carried. However, among all the ribs it is convenient to identify those ribs that should be properly sized because they are structural elements on which concentrated loads will be applied, such as engine pylons or landing gear attachment.

These simple steps are sufficient to sketch the structure. In following iterations, the geometry of the single section elements must be defined, as well as materials and thickness. CAD models can guarantee an optimal support in carrying out these activities.

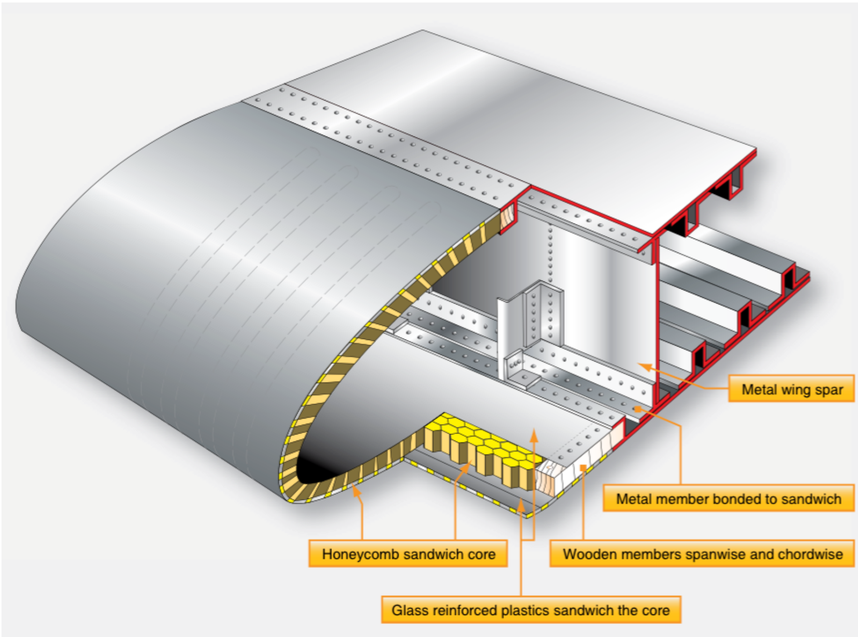


Figure 240: Internal wing structure detail: leading edge construction

7.2.2 Fuselage structure

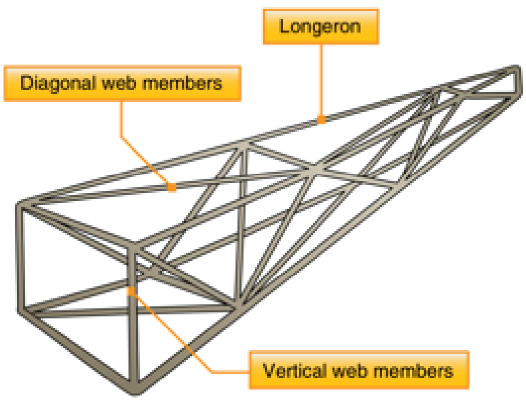
The same procedure followed for the definition of the wing structure, can be used for the structural design of the fuselage. Starting from a qualitative analysis aimed at identifying the main areas of interest, the major functions are derived and as a consequence, required structural elements of the fuselage.

From constructional point of view, three general types of fuselage may be envisaged: truss, monocoque and semimonocoque (Figure 241)

Truss-type fuselage frame is usually constructed if steel tubing welded together in such a manner that all members of the truss can carry both tension and compression loads. This type of structure is mainly used in small aircraft and in case of light performing aircraft, aluminium alloy may also be implied.

The monocoque (Figure 241), or single shell, fuselage relies largely on the strength of the skin in order to carry the primary loads and uses formers, frame assemblies and bulkheads to guarantee the desired shape to the fuselage. At pre-defined intervals, heavy structural elements are located in order to carry concentrated loads. This happens especially where fittings are used to attach other units such as wings, powerplants and empennages. In particular, it has to be noticed that the major objective of structural optimization is trying to connect the highest number of elements to a single heavy structural member in order to minimize the overall aircraft weight. In any case, the biggest problem to be faced with in the case of monocoque structure is maintaining enough strength while keeping the weight within allowable limits.

To overcome the strength/weight problem of monocoque construction, a modification called semimonocoque was developed. It also consists of frame assemblies, bulkheads and formers performing the same functions as in the monocoque configuration but the skin is reinforced by longitudinal members called longerons that usually extend across several frame elements and have the function of helping the skin supporting primary bending loads. Additional elements of this fuselage arrangement are stringers. They are longitudinal elements with a variety of shapes and characterized by a certain level of rigidity but they are chiefly used for giving shape and for skin attachment purposes. Both stringers and longerons together prevent tension and compression from bending the fuselage. Web may be used between longerons and stringers and they can be installed vertically or diagonally.



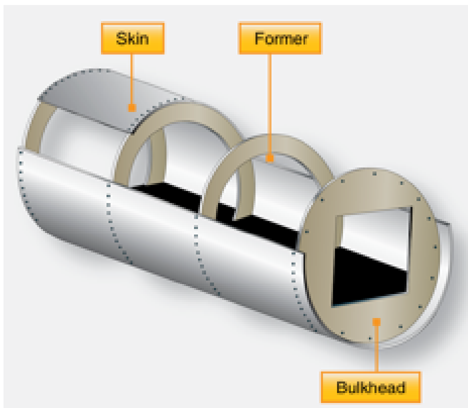


Figure 241: Truss (left) and semimonocoque (right) fuselage configuration

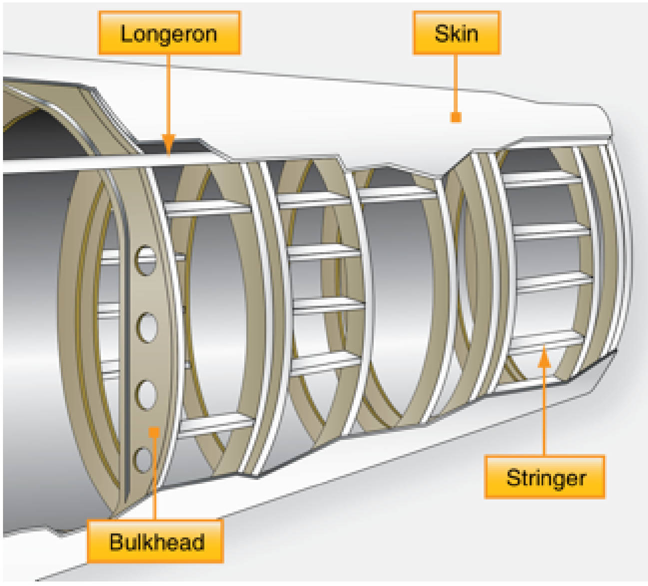


Figure 242: Semimonocoque fuselage structure members

It can be noticed that no one of the aforementioned structural elements can be able to fully support the loads but their capability is hidden behind the integration.

Considering that semimonocoque arrangement (Figure 242) appears to be the most convenient for hypersonic vehicles, the following steps are suggested to sketch the fuselage internal structure. Differently from the wing structure definition, considering the axial symmetry of the body, it is sufficient to work with the top or the lateral view. However, considering that the fuselage structure should be integrated with the rest of the vehicle, the parallel view definition is suggested.

1. Main solid bulkheads should be located at end on the nose section as well as at the end of the passengers' compartment in order to seal the pressurized volume of the fuselage.
2. Additional heavy bulkhead should be envisaged in order to reinforce the structures, in any part in which the skin continuity is not guaranteed, as far as the discontinuities due to openings such as boarding and un-boarding doors, windows or simple doors allowing systems accommodation such as the landing gear.
3. The other bulkhead whose major function will be to contribute to maintain the shape of the fuselage can be located in such a way that

they are closer to each other in that part of the fuselage in which there are more fast changes in fuselage section shape.

4. Conversely, as a first attempt, longerons can be located to a regular distance in order to cover the overall sections diameter and the same can be done for stingers.

The suggested activity flow is also the most efficient way to generate a first fuselage model on a 3D CAD modelling software. Indeed, exploiting the major benefits of parametric CAD modelling, it is possible to define the different structural elements, or at least their location with respect to the reference system and then, once the skin has been generated through a loft and some guidelines, specific geometries for the different elements can be generated.

7.2.3 Wing - Fuselage integration

Wing and fuselage are the two major structural components whose integration is strictly related to the configuration selected at the very beginning of the design process. Indeed, the selection of the wing vertical location, for example, is one of the most impacting decision in terms of wing fuselage integration.

Moreover, during the integration process, aerodynamics, aerothermodynamics and structural issues should be considered. In addition, considerations related to operations, accessibility, production and safety can deeply affect the integration strategy. For example, the need of separating the front part of the fuselage, containing passengers, from the overall configuration, may imply to adopt special integration strategy, allowing the separation. This is only an example of impact of a non-technical area the design process.

The main structural criterion to be followed with the aim of integrating wing and fuselage structures and then, adding all the other subsystems, is the dry mass minimization and this can be reached only whether a single heavy element is able to support the highest number of loads. Thus, from the structural point of view, the optimization in conceptual design phase may be reached properly redefining the internal wing and fuselage structure in such a way that the integration requirements are fulfilled.

7.3 Wing-fuselage longitudinal location

Even if the mutual vertical position of the wing with respect to the fuselage has already been set, the longitudinal location has to be defined yet. This activity is mainly influenced by weigh and balance considerations: indeed, the major requirement to be fulfilled in selecting the proper longitudinal position of the wing with respect to the fuselage is *to guarantee a stable configuration of the overall transportation system*.

At this design stage, when no empennages, nor tail nor canard have been defined yet, in order to guarantee the stability of the configuration, the only design parameter on which the designer can act is the distance between the centre of gravity location and the aerodynamic centre. Indeed, the basic concept of stability is simply that a stable aircraft, when disturbed, tends to return by itself to its original state. However, if the restoring forces are too strong, the aircraft will overshoot the original state and in this case, the problem moves from static stability to dynamic stability. At this design stage, the designer should set up a configuration in which the aerodynamic centre and the CG of the entire vehicle are closed enough.

To do this, the first step is the identification of the wing aerodynamic centre in subsonic conditions. For high-aspect-ratio wing, this point is usually identified at a certain percent of the mean aerodynamic chord (MAC) of the 2D airfoil aerodynamic centre. For most airfoil, this is the quarter-chord point while, when moving to supersonic speeds, the wing aerodynamic centre moves to the 50% of MAC. Literature provides lot of solutions (Raymer, 2012) for a first estimation of the location of this fundamental point, but unfortunately, many of these methods are not applicable in the field of transonic speeds.

For simplicity, a geometrical construction has been adopted here. In order to make this algorithm applicable also to more complex geometrical wing configuration, a preliminary step should envisage the simplification of the wing planform to a trapezoidal shape. For suggestions look at Figure 243

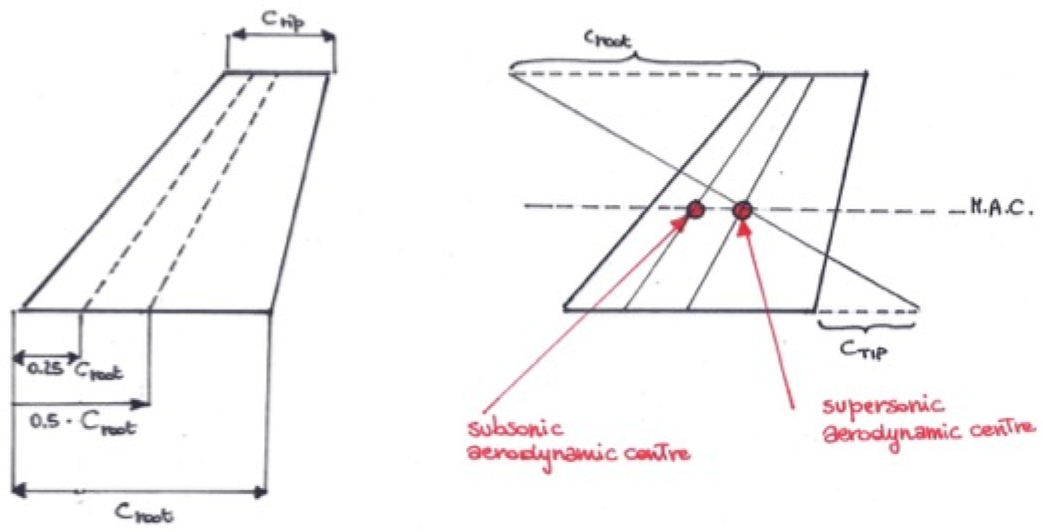


Figure 243: Geometrical construction to identify the location of the wing aerodynamic center.

Once the noticeable points of wing and fuselage have been identified, with an iterative process, the wing should be moved back and forth to identify the most suitable location. As a first attempt, in order to find the optimal location, a graphical construction like the one suggested in Figure 244 can be followed. This graphical representation clearly shows the strict correlation between the aircraft CG and the wing aerodynamic centre.

Once a suitable location has been selected, the designer should verify the feasibility of the geometrical solution from the structural point of view. As highlighted in Figure 245, in case the proposed solution is not optimized from a structural point of view, the wing location should be moved in the closest structurally feasible location and the new difference between CG and aerodynamic centre must be computed.

However, at this stage, not negligible values of this parameter can be accepted taking into account that there are many other elements of the configuration to be placed yet and in some cases, their location and size will impact on the location of the aircraft CG mainly. It is the case of the integration of external propulsion systems, fuel tanks and landing gear. At the end, flight control surfaces may be added in order to guarantee a proper level of stability and control in the different mission phases (Figure 245).

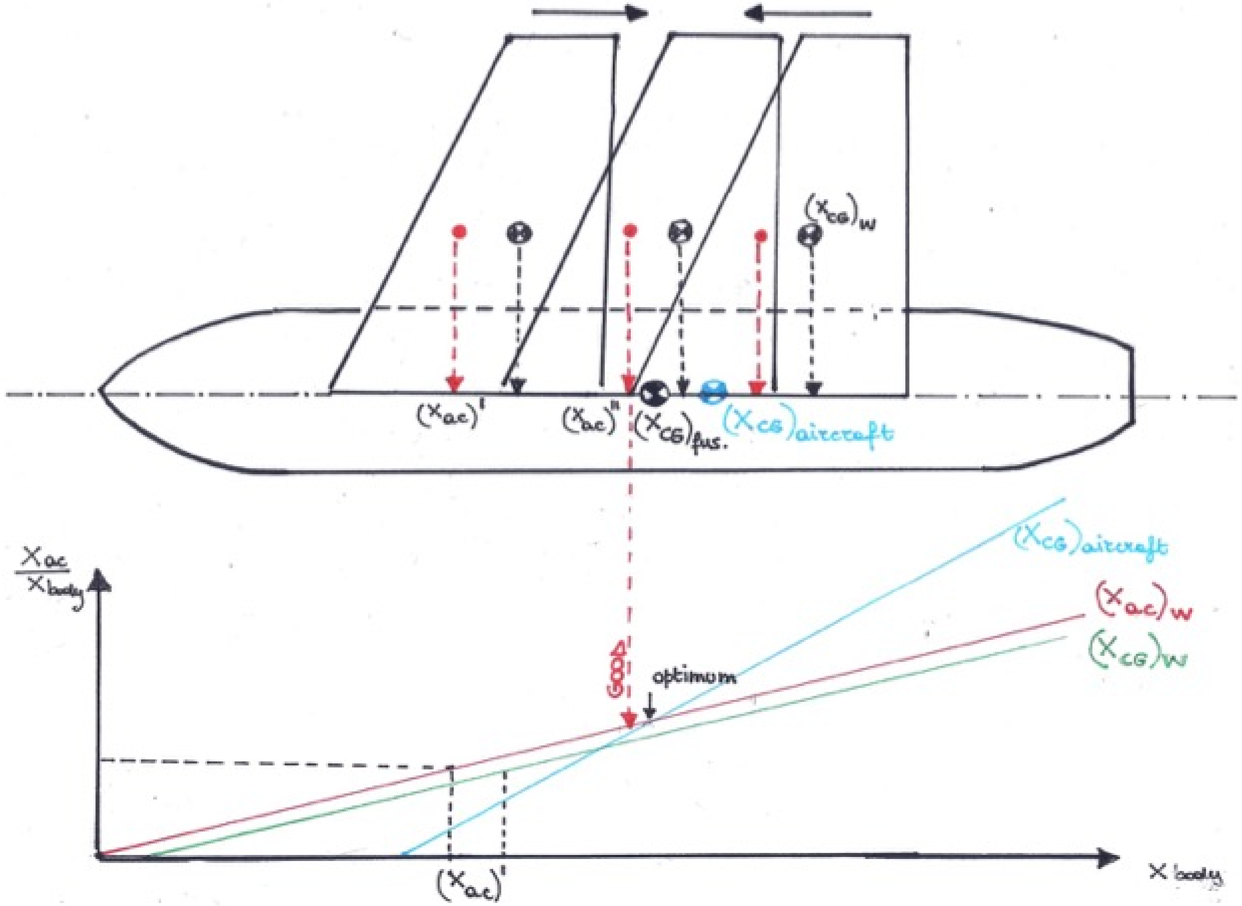


Figure 244: Geometrical construction to identify the optimal longitudinal location of the wing with respect to the fuselage.

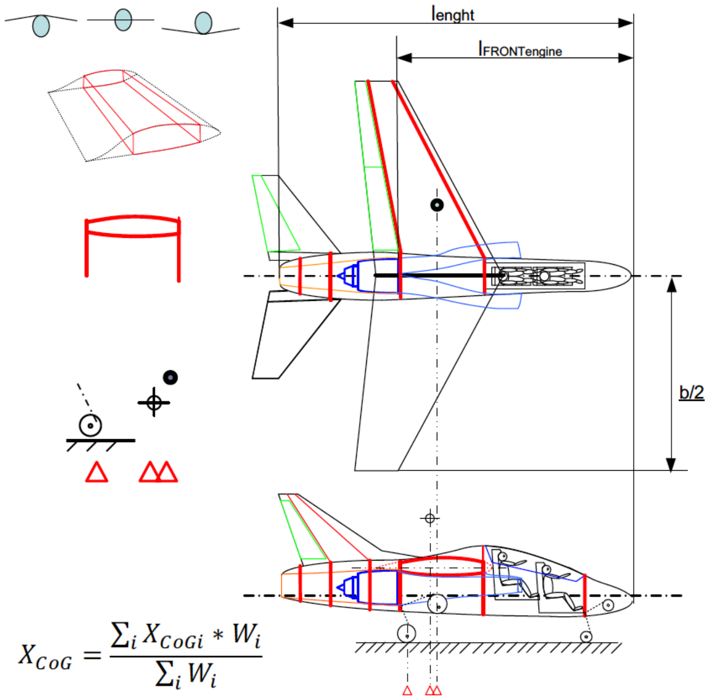


Figure 245: Example of optimal structural wing-fuselage integration

7.3.1 Reference case study: wing and fuselage integration results

This subsection aims at collecting the first complete sketch for the selected reference case study, highlighting the way in which the main subsystems have been integrated within the envisaged external layout.

It has to be noticed that these have only be the very first step that allows generating a complete CAD model and exploit it for several different specific domain analyses (as it is reported in section 7.10).

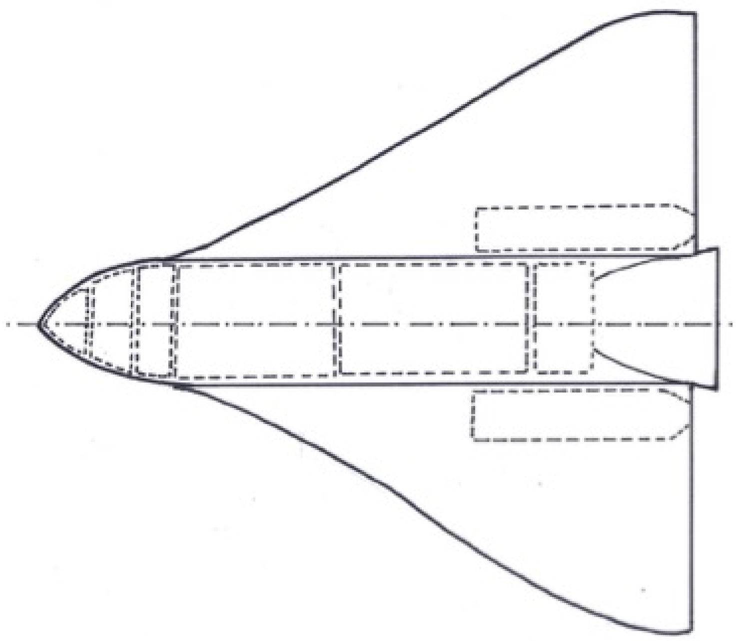


Figure 246: Fuselage-wing integration for the reference case-study

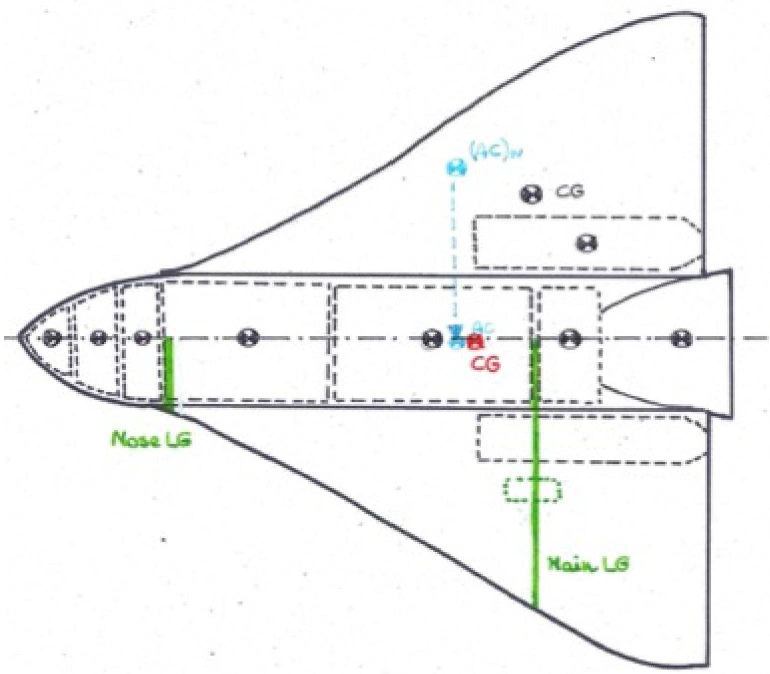


Figure 247: Fuselage-wing integration (CG and AC identification)

7.4 Propulsive system integration

This chapter aims at providing some practical guidelines to face with the problem of propulsion system integration, especially in the case of hypersonic vehicles. In the previous chapter of this Thesis it has been possible noting that the presence of a propulsion system, depending on its location, may have a deep impact on the design of both wing and fuselage. However, besides the impact in terms of configuration layout and sizing, it is important to envisage the proper structural integration of this system into the fuselage + wing configuration. In order to optimize the overall mass of the vehicle, proper structural attachments should be envisaged. In the selection of the proper location and subsequent integration of the propulsion system, the following considerations should be taken into account:

- If dealing with air-breathing engines, the inlet and exit nozzles should be properly considered during the structural integration;
- In case of multi-engines configuration, the setting must be such that the loss of one power source does not affect the handling of the aircraft in a catastrophic way.
- In case of VTOL capabilities, or in case of specific performance required at high angle of attacks, the inlet and outlet location should be properly selected in order to guarantee the fulfilment of the requirements during all the mission phases.
- In case rotating parts are present, proper safety consideration shall be taken onto account avoiding accidentally contacts of debris with the crew and passengers' compartments in case of failure. In this case, the problem can be mitigated with the adoption of reinforced structures or with proper location. In this last case, it is also important to avoid snow-ball effects.

Taking a close look at the hypersonic domain, it is possible to notice that the need of propulsion systems able to guarantee performance in a so wide range of Mach numbers and altitude, forced the scientific and technical community to find integrated solutions to avoid the co-presence of two or three different propulsion subsystems each with a limited field of application. The currently on-going propulsion integration may allow noticeable mass savings and a drastic reduction of integration issues. In addition, considering the problems related to the integration of propulsion systems into the airframe, avoiding aerodynamics and aerothermodynamics problems, peculiar strategies such as the "propulsive fuselage" might be envisaged. In this case, the aft part of the fuselage is entirely dedicated to host the propulsion system, perfectly integrated within the fuselage main body. Complementary, other studies are looking towards "propulsive crown". These innovative configurations are not only under-study for the case of hypersonic, but also in the traditional aeronautics, due to promising benefits, also in terms of aeroacoustics noise reduction as well as pollutant emissions.

7.5 Fuel tanks integration

The integration of fuel tanks has been dealt with in different previous chapters. As far as their structural integration is concerned, different possible strategies may be adopted. However, thinking about hypersonic vehicles, the most interesting and promising solution will be the direct integration within the airframe, allowing mass saving and maximizing the internal available volume (Figure 248). External tanks may be added but the only possibility of successfully completing a hypersonic mission is to detached these tanks after fuel completion but this is against the concept of reusability aimed at by the different aerospace domains interested in hypersonic missions.

In case integrated tanks will not be used, the problem of defining the optimal number of propellant tanks may be considered. The number and the size of tanks will be the result of a trade-off considering two requirements mainly (Figure 249):

- The propellant tanks shall be designed in such a way that the depletion strategy may ensure the minimum CG shifting over the entire mission.
- The propellant tanks shall be designed pursuing the minimum dry mass objective

Unfortunately, the easiest way to fulfil the first requirement is to increase the number of tanks to enhance the fuel depletion but this strategy will incredibly augment the dry weight of the vehicle. Intermediate solutions may be identified, taking also into account the possibility of introducing CG shifting control strategy in a Corcorde-like fashion.

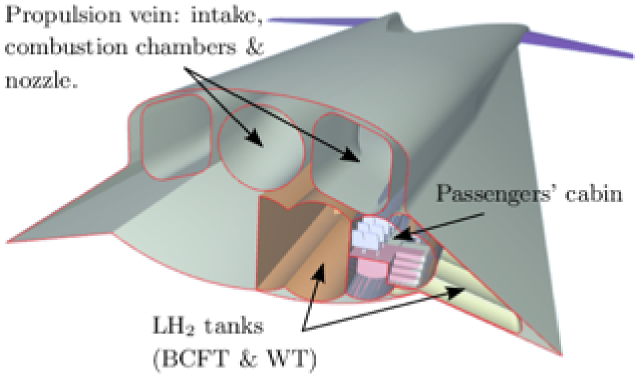


Figure 248: LAPCAT MR2 integrated propellant tanks (Villace, 2015)

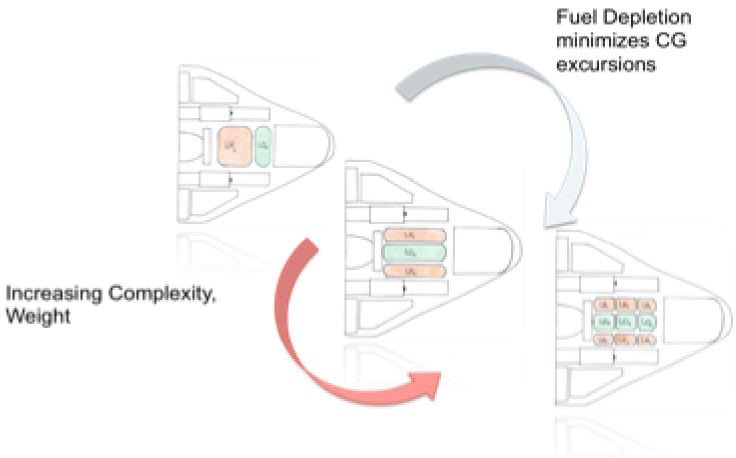


Figure 249: Example of trade-off for the selection of the optimal number of tanks

7.6 Landing Gear

The landing gear is another fundamental system for the overall configuration and its location should be properly selected in order to guarantee the aircraft stability when on ground but also the possibility to perform take-off and landing operations (Raymer, 2012) (Sadraey, 2012).

Before moving to the selection of its most suitable location, it is important to envisage a proper layout and sizing for this element, thinking in particular to the specific needs of the missions dealing with hypersonic.

For this reason, following a systems engineering approach, the major functionalities to be pursued are listed. Then, a list of requirements is elicited and categorized depending on the Area of Interest. From this analysis. The results of this analysis are summarized here below.

Main functions to be allocated to the Landing Gear System:

- To support the aircraft when on-ground
- To guarantee aircraft stability when on ground
- To support the aircraft during taxi phase
    - To support static loads during taxi phase
- To support the aircraft during take-off phase
    - To allow aircraft rotation
    - To sustain structural loads during take-off
- To support the aircraft during landing
    - To contribute to the aircraft deceleration
    - To sustain structural loads during landing
    - To absorb shocks during landing
    - To sustain thermal loads during landing
- To guarantee aircraft manoeuvrability during taxi phase
- To guarantee aircraft manoeuvrability during take-off phase
- To guarantee aircraft manoeuvrability during landing phase
- To provide separation of the airframe from ground

Requirements:

Stability and Control

- *The landing gear shall be able to support the aircraft when on-ground*
- *The landing gear shall be able to guarantee the aircraft stability when on-ground*
- *The landing gear shall guarantee aircraft manoeuvrability during taxi phase*
- *The landing gear guarantee aircraft manoeuvrability during take-off*
- *The landing gear guarantee aircraft manoeuvrability during landing phase*

Logistics and Operations

- *The landing gear shall allow rotation during take-off phase*
- *The landing gear shall contribute to the aircraft deceleration*
- *The landing gear provide separation of the airframe from ground*
- *The landing gear shall prevent tail to hit the ground.*
- *The landing gear shall guarantee proper propeller ground clearance*

Structure and Mechanisms

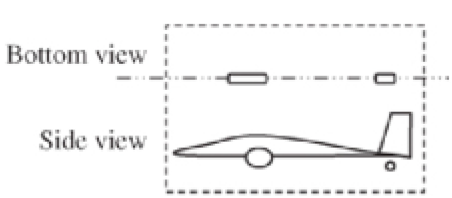
- *The landing gear shall support static loads during the taxi phase*
- *The landing gear shall support static loads during take-off*
- *The landing gear shall support static loads during landing*
- *The landing gear shall absorb shock during landing*
- *The landing gear shall sustain thermal loads during landing*

Then, in the next subsections, correlations between requirements and design parameters can be established guaranteeing a proper level of traceability.

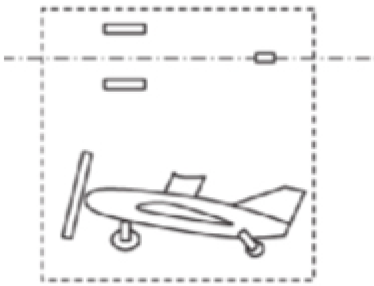
7.6.1 Landing gear configuration

The existing landing gear configurations are reported in Table 109 with a small description for each, highlighting the possible field of applications.

Table 109: Summary of landing gear configurations

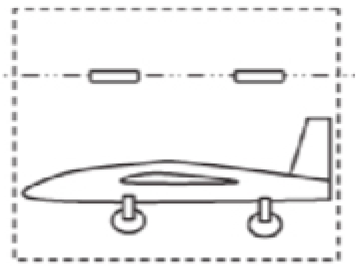
Configuration	***Comments***
Single Main 	The single main gear configuration is the simplest that can be envisaged and it can be adopted for sailplanes. For more complex, larger and heavier aircraft, this configuration is no more applicable. The most common configuration the wheel is usually installed forward of the centre of gravity or aft with the addition of a skid under the cockpit.

Taildragger



This configuration has two main wheels forward the CG and a smaller auxiliary wheel at the tail. Due to the wide exploitation of such configuration during the first 40 years of aviation history, this configuration is also addressed with the term “conventional landing gear”. Today, this landing gear layout survives in some sport airplanes. With respect to the previous configuration, this alterative provides a higher propeller clearance from ground, allowing to generate more lift for rough field operations. Conversely, this configuration is inherently unstable impacting on the handling qualities in both take-off and landing phases.

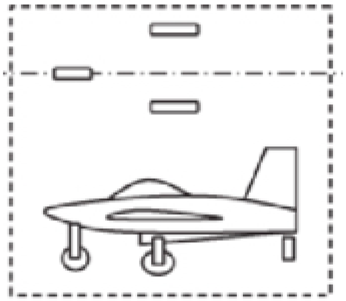
Bicycle



This configuration has two main wheels, one for and one aft the CG but to prevent the aircraft from tipping sideways, small outrigger wheels shall be added. However, whether the rear wheel is placed so far behind the CG, difficulties in carrying out take-off manoeuvre shall be overcome exploiting high-lift devices allowing to increase the aircraft performance at low speeds and low angles of attack. Conversely, the small wheel-track allows the exploitation of this configuration in for narrow fuselage aircraft , or in special cases, such as the Harrier, where the engine and nozzles location has been located where normal landing gear would be installed.

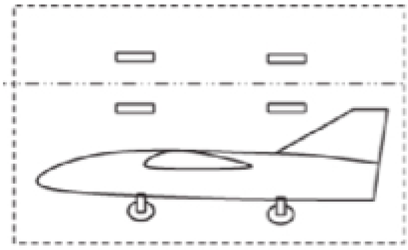
Tricycle

This is the most widely exploited landing gear arrangement, consisting pf two main wheels behind the CG and an auxiliary wheel forward of the CG. This configuration guarantees the aircraft stability on-ground with



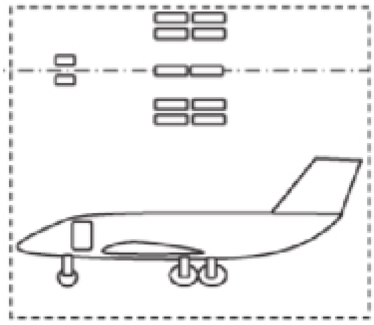
noticeable benefits in terms of take-off and landing operations. Moreover, this configuration guarantees the satisfaction pilot forward visibility requirements and the flatness of the crew and passengers compartments floor.

Quadricycle



The quadricycle gear is much like bicycle with the wheels located on the side of the fuselage. As well as the Bicycle configuration, this type of landing gear arrangement requires flat take-off and landing runways. As advantage, the quadricycle configuration allows the very low floor and thus it can be a desired configuration for cargo aircraft.

Multi-bogey



The multi-bogey configuration is a landing gear with multiple gears and are useful in case of increasing mass of the aircraft. It is usually very stable on type ground and also during taxi operations. On the opposite, this configuration results to be the most expensive and complex as far as production and maintenance are concerned.

However, considering hypersonic vehicles and their envisaged characteristics, Tricycle arrangements appears to be the most promising solution. In order to rationalize the selection process of the most suitable landing gear configuration for a hypersonic vehicle, a trade-off has been carried out focusing on the following Figures of Merits, directly extrapolated from the comments reported in Table 110.

- Costs

- Manufacturability
- Take-off and landing operations
- On-ground stability
- Taxiing stability

As far as the evaluation criteria is concerned, a traditional scale from 1 to 10 has been used, while the weights expressing the impact of each single figures of merit on the selection have been allocated on percentage base (Figure 249).

Table 110: Landing gear configuration trade-off results

	Weigth	Single Main	Bicycle	Tail-dragger	Tricycle	Quadricycle	Multi Bogey
Cost	0,2	9	7	6	4	2	1
Manufacturability	0,2	3	4	5	7	9	1
Take-off and Landing Operations	0,2	3	4	6	10	5	8
On-ground stability	0,2	1	2	7	9	10	8
Taxiing stability	0,2	2	3	1	8	10	9
		3,6	4	5	7,6	7,2	5,4

Thus, considering the winning solution, in the next subsection, the tricycle arrangement alternative is detailed, suggesting a sizing algorithm and landing gear integration procedure.

7.6.1.1 Landing gear configuration selection for the reference vehicle

As far as the reference vehicle is concerned, in order to select the most adequate landing gear configuration, proper weighting strategy should be adopted to initialize the trade-off in order to respect all stakeholders' expectations. In particular, considering that among the most impacting mission requirements there is the one prescribing VTOL operations, the stability during take-off and landing phases is essential and for this reason, the trade-off has been performed, with a different weighting strategy of the FoMs. In any case, as it is possible noticing in Table 111, the Tricycle configuration results to be the winner.

Table 111: Landing gear configuration trade-off results for the reference case study

	Weigth	Single Main	Bicycle	Tail-dragger	Tricycle	Quadricycle	Multi Bogey
Cost	0,125	9	7	6	4	2	1
Manufacturability	0,125	3	4	5	7	9	1
Take-off and Landing Operations	0,5	3	4	6	10	5	8
On-ground stability	0,125	1	2	7	9	10	8
Taxiing stability	0,125	2	3	1	8	10	9
		3,375	4	5,375	8,5	6,375	6,375

7.6.2 Landing gear sizing

Before starting with the sizing process, it is important to identify the major sizing parameters for the selected configuration and try to allocate the previously derived requirements onto them, guaranteeing a proper traceability. In this case, considering that we are working with relationships between requirements and configuration elements, it is correct to speak about external traceability. The starting list of requirements is the one provided in the previous subsection However, additional specification or performances related to the growing information about the mission profile (that is evolving during the aircraft design activities) allows to generate a new list of requirements. In addition, Figure 250 summarizes the major activity flow here suggested for an optimal landing gear design and integration.

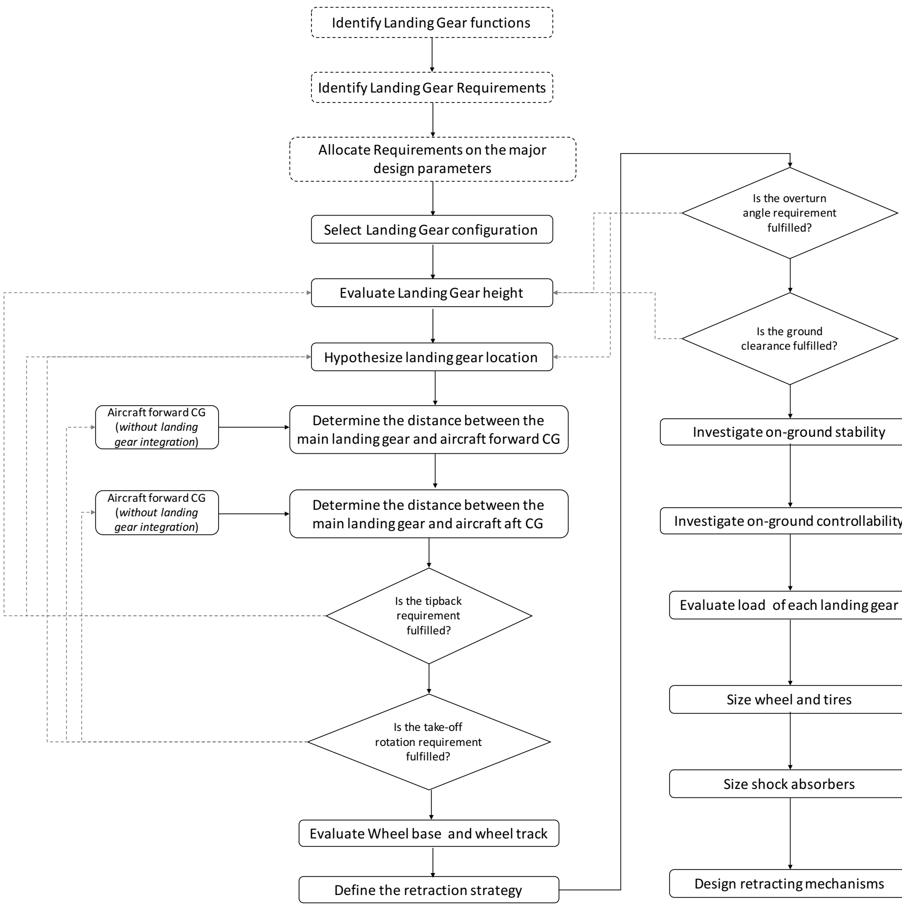


Figure 250: Flow-chart summarizing the landing gear suggested design and integration process

7.6.2.2 Landing Gear Height

Among the main design parameters, there is the Landing Gear Height that can be defined as the distance between ground and the conjunction between the main landing gear strut and the airframe. Even if dealing with conceptual design phase, it is important noticing that the height of the landing gear might be shorter when the aircraft is on ground depending on spring deflection or oleo compression due to the aircraft weight acting on the leg. For this reason, it is common to define this parameter as the height of the landing gear when the aircraft is on-ground and the fuselage is in the horizontal position.

This is a list of requirements that may have a direct impact on the definition of the landing gear height.

- The landing gear height shall guarantee proper clearance during taxi operations
- The landing gear height shall provide rear fuselage clearance during take-off rotation
- The landing gear height shall prevent tipback phenomenon (tail strike)
- The landing gear height shall prevent overturn
- The landing gear height shall satisfy loading and unloading operation requirements.

As far as the required clearance is concerned, different suggestions may be envisaged for the different aircraft categories, of the different aircraft elements with respect to ground. Suggestions coming from CS are summarized in Table 112. Looking in particular to the Take-Off rotation phase, it is important to properly derive the height of the landing gear in such a way that tail strikes are prevented. Translating these thoughts into mathematical equations, the following expressions may be used:

$$\alpha_C \geq \alpha_{TO}$$

where the clearance angle α_C can be evaluated as it follows:

$$\alpha_C = \tan^{-1}\left(\frac{H_f}{AB}\right)$$

Table 112: Clearance suggestions

Aircraft Component	*Clearance [m]*
Fuselage	0,2 – 1,2
Rear Fuselage	0,2 – 0,5
Wing	0,2 – 1,5
Turbofan/turbojet engine	0,5 – 1,5
Propeller (landplane)	0,2 - 1
Propeller (seaplane)	1 – 2
Store/fuel tank/ antennas and other	0,2 – 0,6

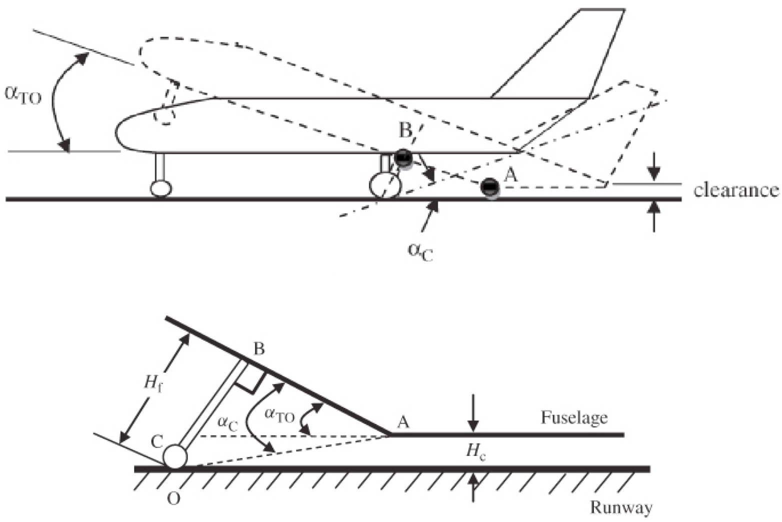


Figure 251: Graphical evaluation of clearance requirement

This means that if the clearance angle (α_C) is less than the aircraft rotational angle (α_{TO}) during take-off, the fuselage will strike the ground. Otherwise, there will be clearance between the fuselage and the ground, preventing from any damages to the fuselage.

Landing Gear Wheel Base

Another very important parameter of the configuration is the wheel base that can be defined as the distance between the main and the secondary landing gear measured along the longitudinal axis. It plays a fundamental role in the load distribution between main and secondary landing gear as well as the ground controllability of the vehicle controllability and ground stability. Indeed, the following list of requirements must be satisfied:

- The wheel base shall guarantee proper load distribution among between the main and secondary landing gear.
- The wheel base shall be properly defined in order to face with possible CG shifting.
- The wheel base shall be properly defined in order to sustain static and dynamic loads experienced during take-off and landing.
- The wheel base shall be sized in order to guarantee adequate aircraft controllability when on ground.
- The wheel base shall be sized in order to guarantee aircraft stability when on ground.

Aiming at satisfying the first two requirements, the following set of equations, based on the equilibrium in a static loading condition may be exploited. Even if at conceptual design stage, possible CG shifting shall be taken into account. For this reason, simple set of equations may be rewritten allowing taking into account the minimum and the maximum loading percentage of the main and secondary landing gear.

$$\begin{cases} F_{s_{max}} = \dfrac{B_{s_{max}}}{B} W \\ F_{m_{max}} = \dfrac{B_{m_{max}}}{B} W \\ F_{s_{min}} = \dfrac{B_{s_{min}}}{B} W \\ F_{m_{min}} = \dfrac{B_{m_{min}}}{B} W \end{cases}$$

where:

F_s is the percentage of static load acting on the secondary landing gear;

F_m is the percentage of static load acting on the main landing gear;

B_m is the relative distance of the main landing gear with respect to the aircraft CG

B_s is the relative distance of the secondary landing gear with respect to the aircraft CG

Thinking to possibility of withstanding dynamic load, mainly due to the aircraft acceleration and deceleration experienced by the aircraft during take-off and landing phases, the following set of equation may be used:

$$\begin{cases} F_{s_{dyn}} = |a_L| \dfrac{W\ H_{CG}}{gB} \\ F_{m_{dyn}} = a_T \dfrac{W\ H_{CG}}{gB} \end{cases}$$

where

$F_{s_{dyn}}$ is the dynamic load acting on the secondary landing gear;

$F_{m_{dyn}}$ is the dynamic load acting on the main landing gear;

W is the Maximum Take-Off Weight;

H_{CG} is the distance of the aircraft CG from the terrain;

a_L is the braking deceleration;

a_T is the take-off acceleration

g is the gravitational acceleration

Thus, the maximum static and dynamic loading can be exploited in order to properly define and size the wheel base.

Landing Gear Wheel Track

The wheel track is another fundamental design parameter of the landing gear that can be defined as the distance between the most left and the most right gears looking the aircraft from the front view. Its definition can be guided by the following requirements:

- The wheel track shall guarantee adequate vehicle lateral control when on-ground.
- The wheel track shall guarantee adequate vehicle lateral stability when on-ground.
- The wheel track shall guarantee the aircraft structural stability.

When performing the sizing of the wheel track (T), the minimum allowable value shall satisfy the overturn angle requirement (i.e. the lateral control) (Figure 252) while the maximum allowable value must satisfy the structural integrity requirements.

In order to ensure ground controllability: $T > 2\frac{F_C H_{CG}}{mg}$

In order to ensure ground stability: $T > 2\frac{F_W H_C}{mg}$

In order to ensure structural integrity: $T < \sqrt[3]{\left[\frac{48\, EI\, B\, y_{max}}{WB_{s_{max}}}\right]}$

Where:

F_C is the centrifugal force;

F_W is the cross-wind force;

H_C is the height of the centroid of the aircraft from the ground (Figure 253);

y_{max} is he maximum deflection (Figure 254);

E is the modulus of elasticity;

I is the second moment of inertia of the beam area used to simplify the structure.

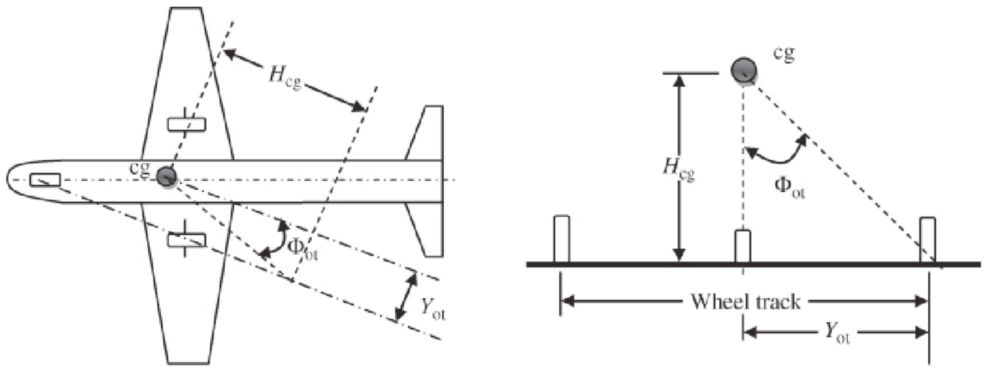


Figure 252: Graphical evaluation of overturn requirement

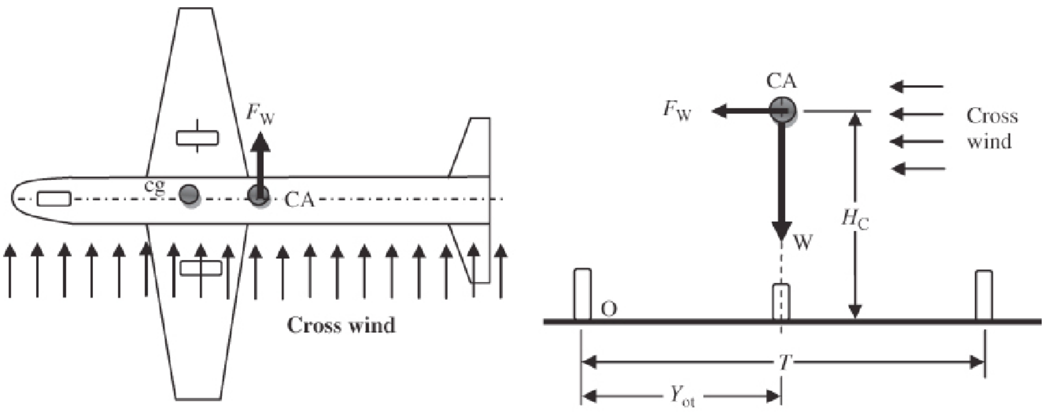


Figure 253: Graphical evaluation of ground stability requirement

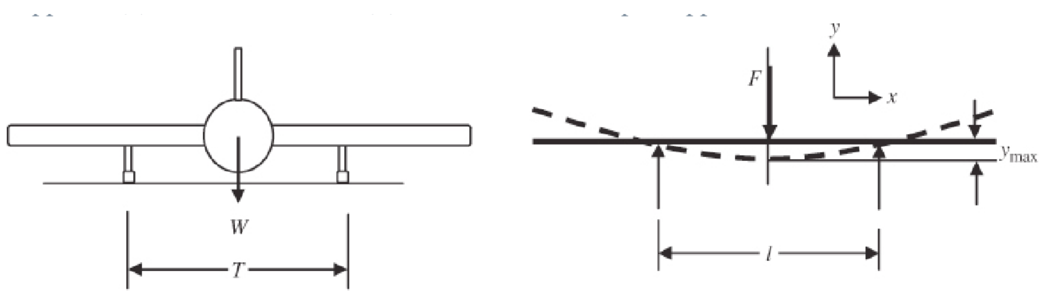


Figure 254: Graphical evaluation of structural stability requirement

7.6.3 Landing gear location

The landing gear location is mainly related to the CG location. In particular, for a tricycle arrangement, the integration of such important subsystem with respect to the overall vehicle configuration is strictly dependent on the two extreme expected aircraft CG locations. Moreover, the fulfilment of the tipback and tipfoward requirements as well as the take-off rotation requirement (Figure 255). Following these considerations, this list of requirements can be elicited:

- The landing gear shall be properly located to avoid the aircraft aft fuselage to hit the ground during take-off rotation manoeuvre.
- The landing gear shall be properly located to ensure an adequate level of manoeuvrability to the aircraft during take-off manoeuvre.

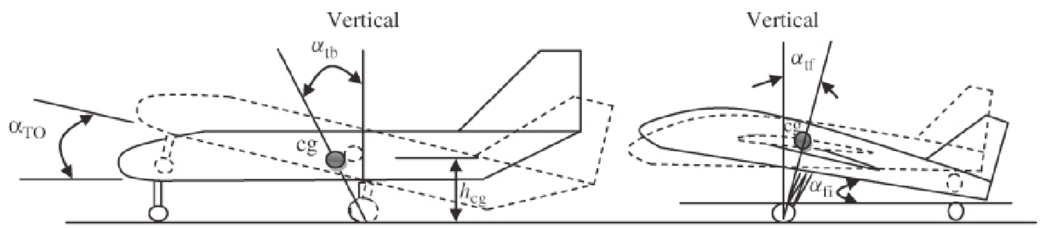


Figure 255: Graphical evaluation of tipback angle, tipforward angle and take-off rotation requirements

As far as the tipback requirement is concerned, to prevent tipback (hypothesizing a tricycle configuration) the tipback angle (as defined in Figure 255) shall exceed of at least 5 degrees the take-off rotation angle, where 5 deg is suggested from literature as a safety design margin to cover a series of possible design and operation uncertainties. Considering that typical take-off rotation angles are stand between 10 deg and 15 deg, tipback angle must be equal to or greater than 15-20 deg. Two possible design strategy may be adopted to increase the tipback angle:

- Reduce the landing gear height
- Move aft the main landing gear

Considering these requirements, the difference $(x_{mg} - x_{cg})$ results to be the maximum allowable distance for the main landing gear location and exploiting the equilibrium equations.

Considering the tipforward angle, similar considerations can be carried out. Of course, this angle can be considered only in case of special aircraft configurations where a tail-wheel landing gear architecture has been envisaged. In this case the angle shall exceed the fuselage inclination angle of at least the same 5 deg selected for safety reasons.

Eventually, the location of the landing gear with respect to the aircraft centre of gravity shall fulfil the take-off rotation requirement. In particular, the distance of the main landing gear to respect to the aircraft nose can be evaluated solving the equilibrium equation mathematically describing the take-off.

$$x_{\mathrm{mg}} = \frac{I_{yy_{\mathrm{mg}}}\ddot{\theta} - D\left(z_D - z_{\mathrm{mg}}\right) + T\left(z_T - z_{\mathrm{mg}}\right) - M_{\mathrm{ac_{wf}}} - ma\left(z_{\mathrm{cg}} - z_{\mathrm{mg}}\right) - W x_{\mathrm{cg}} + L_{\mathrm{wf}} x_{\mathrm{ac_{wf}}} + L_{\mathrm{h}} x_{\mathrm{ac_{h}}}}{L_{\mathrm{wf}} + L_{\mathrm{h}} - W}$$

where the different variables are graphically described in (Figure 256)

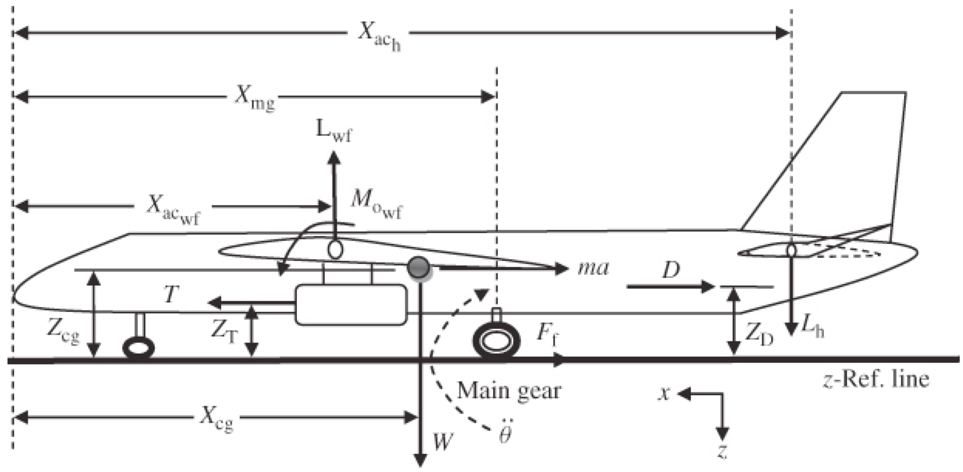


Figure 256: Graphical representation of the variables required to the evaluation of the equilibrium in take-off rotation

7.6.4 Landing gear retraction/extraction

Focusing on hypersonic vehicles, one of the most affecting requirements on the landing gear is for sure its need of being retracted avoiding any additional drag source. We have already seen in the previous chapters that the landing gear can be hosted in different parts of the aircraft, wing, fuselage and so on.

Indeed, it is important to notice that a bad location of the retracted landing gear can affect the overall aircraft layout, its aerodynamic characteristics but it can also interrupt structure provoking additional weights due to the need of reinforcements or the internal fuel volume.

Depending on the landing gear attachment and on the aircraft configuration, one of the following retraction strategies may be used. Please, notice that, configurations presenting external pods or too big fairing may be not suitable for aircraft aiming at performing hypersonic missions.

The identification of the proper location to store the landing gear and to evaluate proper extraction and retraction sequences is one of the major advantages coming from the introduction of a parametric CAD model within the envisaged MBSE tool chain. The specific example of landing gear design, integration and simulation is exploited in later-on in this Chapter to show the advantages of the envisaged MBSE tool-chain and its support for integration purposes.

7.7 Empennages Design and FCS integration

The integration of the Flight Control System for a hypersonic vehicle is a very challenging activity. Indeed, these vehicles should be equipped in such a way that they are able to guarantee a safe flight during the overall mission profiles, i.e. both during atmospheric phase and when in space. This means that the designed control surfaces should be able to guarantee the right balance between stability and controllability. In addition, proper handling qualities should be guaranteed in order to ensure to the pilot and flight participants adequate comfort. The following flowchart shows the main activities to be carried out to design and integrate a Flight Control System. The starting step is the identification of stability and controllability requirements and specific requirements related to the proper mission profile to be performed. In the list of requirements, handling qualities should also be considered especially in following design iterations. As far as the sizing is concerned, it is mainly affected by the control surfaces selected configuration and gravity centre ranges. Then, the sizing process is carried out in parallel for the control along the three major axes and eventually, detailed investigation of the mutual interactions among the sized surfaces should be performed starting new design cycles aimed at design optimization.

7.7.1 Empennages and FCS Requirements

This subsection collects the major requirements allowing a proper design of the aircraft tail and Flight Control Surfaces. It is interesting to notice that this set of requirements is also applicable to non-traditional configurations, such as tailless ones or those with a canard surface. In the following list, the logical subject used is the noun "empennages". However, depending from the selected strategy, either fixed and movable part may satisfy them.

Stability

- The empennages shall guarantee the aircraft longitudinal trim
- The empennages shall guarantee the aircraft directional trim
- The empennages shall guarantee the aircraft lateral trim
- The empennages shall guarantee the aircraft longitudinal stability
- The empennages shall guarantee the aircraft directional stability
- The empennages shall guarantee the aircraft lateral stability

Handling Qualities

- The empennages shall guarantee proper handling qualities, ensuring adequate pilots comfort levels
- The empennages shall guarantee passengers comfort

Safety and airworthiness

- *The empennages shall guarantee a proper level of safety during all the mission phases.*
- *The empennages shall prevent from stall*
- *The empennages shall guarantee spin recovery.*

Development, Production and Logistics

- The empennages layout shall have a low impact on the overall aircraft manufacturing process
- The empennages size shall be the minimum impact on on-ground infrastructure
- The empennages development and production costs should be minimized.

Operations

- *The empennages shall guarantee proper pilot view.*
- *The empennages shall guarantee stealth characteristics.*

7.7.2 Empennages configuration alternatives

The very first step following the requirements elicitation process is the identification of the best empennages configuration option. In the following Figure, the main empennages configurations have been presented.

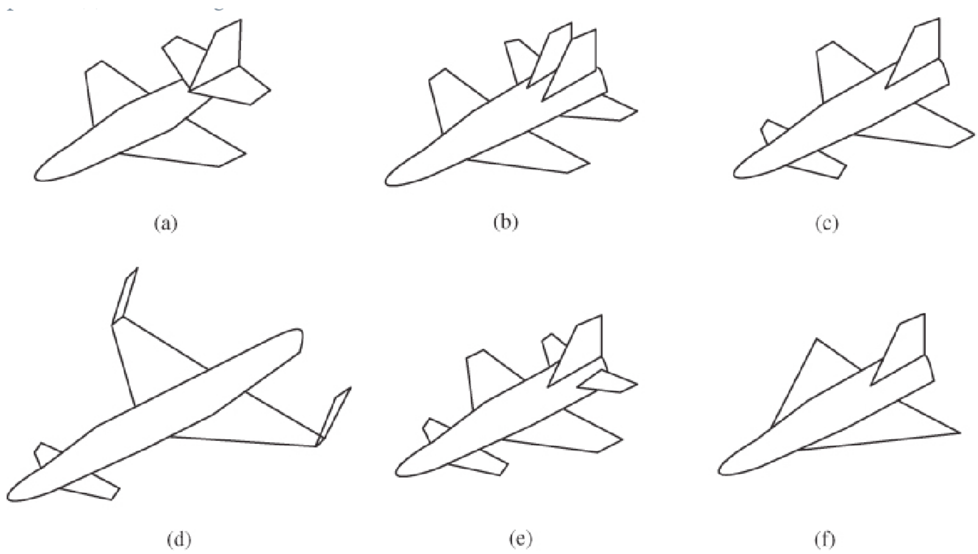


Figure 257: Empennages configuration alternatives

Then, in case a tail configuration is selected, different alternatives exist for the tail. As in the previous case, Figure 258 helps in summarizing the major available alternatives, providing some comments and suggestion for the exploitation.

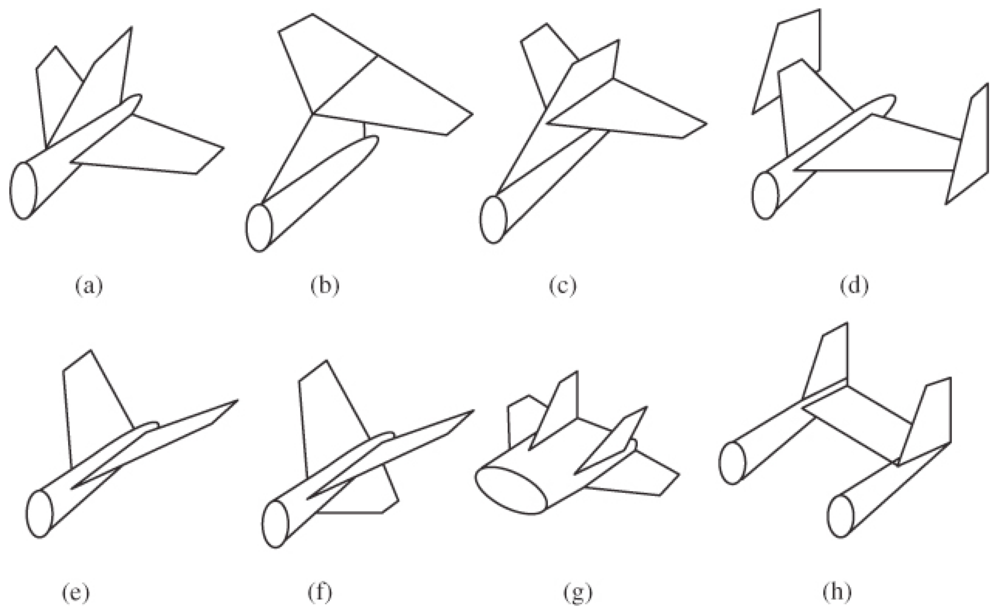


Figure 258: Tail configuration alternatives

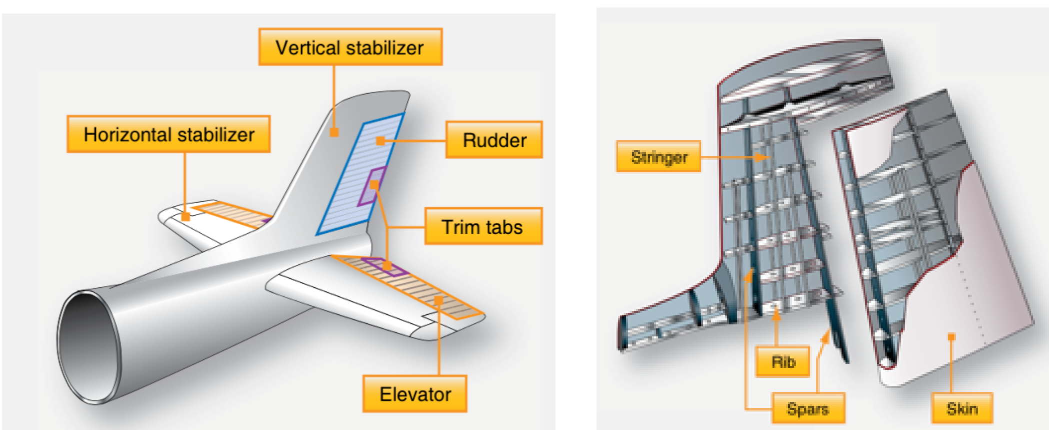


Figure 259: Tail configuration nomenclature and main structural members

It is also important to notice that in some cases, the best options are not possible to be integrated with the already developed aircraft configuration. Among all the possible alternatives, the canard configuration is the most distant from the conventional one. Its presence has a great impact especially in the form of the equilibrium equations to be exploited to represent the behaviour of the complete aircraft configuration.

Following the same approach used all along this Thesis, the best option for each case can be identified on the basis of properly carried out trade-off analyses that, starting from the requirements list already elicited and reported in this section, may allow to derive ad-hoc FoMs to evaluate the best option as far as configuration is concerned.

7.7.3 Optimum tail arm definition

Once the empennages configuration has been selected, the optimum tail arm shall be sized. This parameter is very important because it serves as the arm for the tail pitching moment about the aircraft CG to maintain the longitudinal trim. Before starting the sizing process, it is important to notice that the tail arm is strictly related to the tail area, responsible for the generation of the tail lift. In order to guarantee stability, as the tail arm is increased, the tail area must be decreased, and vice versa.

Complementary, it is possible to identify the major impacting requirements that could be added to the previous list of requirements having an impact on empennages and related control surfaces.

- The tail configuration shall allow minimizing the aircraft weight
- The tail configuration shall allow minimizing the overall aircraft drag
- The tail configuration shall allow minimizing the aircraft wetted area.

Analysing these requirements, it is clear that tail arm and tail area are two parameters whose size shall be properly traded. Indeed, for example, as the horizontal tail arm increases, the contribute of the fuselage to the wetted area increases but the contribute of the tail decreases. For this reason, the major goal of a designer during the conceptual design phase is to determine the optimum combination of tail arm and area that will allow to minimize drag, minimizing the total wetted area of the aft portion of the aircraft.

From a conceptual design standpoint, the wetted area of the aft aircraft can be simplified as the summary of two different contributes: the wetted area of the aft fuselage and the wetted area of the horizontal tail. The first contribute can be evaluated as the lateral surface of a cone with a diameter equal to the diameter of the fuselage at the end of the passengers' compartment or at the end of the system compartment (where present), and a length equal to the length of the aft fuselage section. Complementary, the second contribute can be hypothesized to be twice the empennages planform area.

In order to evaluate the optimal tail arm length that will allow minimizing the zero-lift drag of the aircraft, it is possible to differentiate the wetted area of the aft fuselage part with respect to the tail arm and set this derivative equal to zero. Solving this equation, the following useful equation has been obtained:

$$l_{opt} = k_C \sqrt{\frac{4\, C\, S\, V_H}{\pi\, D_f}}$$

Where:

C is the MAC;

S is the wing surface;

V_H is the volume tail coefficient;

D_f is the fuselage diameter.

Please, notice that a design correction factor $k_C$has been introduced in order to account for the fact that the tail arm may not exactly coincide with the length of the aft fuselage section. Indeed, depending on the layout of the ending part of the fuselage, these two lengths may differ a little bit.

In this equation, there is an important parameter called volume coefficient equation, that can be defined as:

$$V_H = \frac{l}{C}\frac{S_h}{S}$$

where the variables are defined in Figure 260.

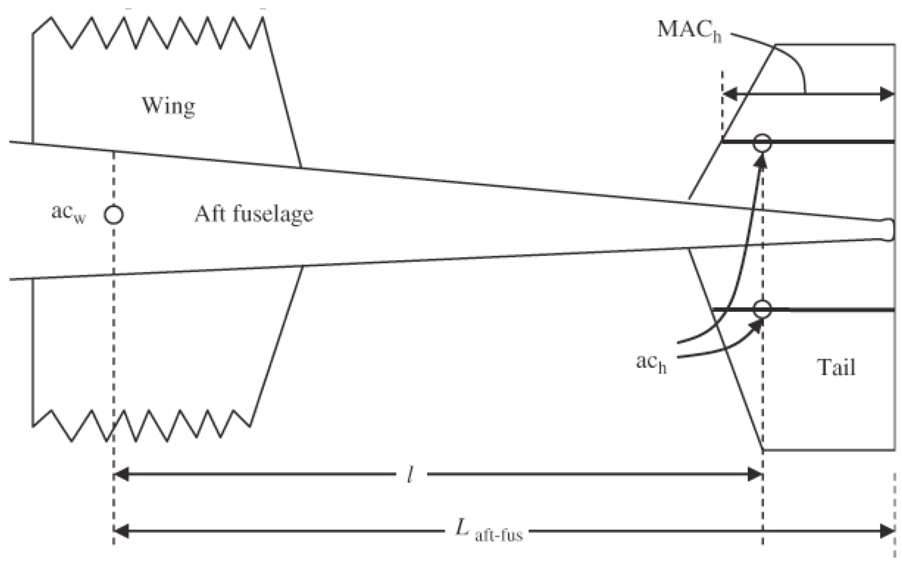


Figure 260: Tail-arm definition

The same equations can be applied to determine the vertical tail volume coefficient. Alternatively, following the suggestions of the following table, these parameters can be estimated on the basis of statistical data (Table 113).

Depending on the need of the aircraft of being able to satisfy all the previously listed requirements all along the mission, during which different flight conditions may be encountered, with variations in terms of aircraft cg location (mainly due to the fuel and propellant depletion), weight, flight altitude and speed, a change in the horizontal tail lift is required. However, considering that the tail area and the airfoil section cannot be changed during a mission, the only possibility is to vary the angle of attack of the surface. In order to satisfy this need, three different strategies may be adopted:

1. Fixed horizontal tail
2. Adjustable tail
3. All-moving tail

Table 113: Horizontal and vertical volume tail coefficient suggestions

Aircraft Type	*Horizontal tail volume coefficient*	*Vertical tail volume coefficient*
Glider	0,6	0,03
Home-built	0,5	0,04
GA single prop-driven engine	0,7	0,04
GA twin prop-driven engine	0,8	0,07
GA with canard	0,6	0,05
Agricultural	0,5	0,04
Twin turboprop	0,9	0,08
Jet trainer	0,7	0,06
Fighter aircraft	0,4	0,07
Fighter with canard	0,1	0,06
Military transport	1	0,08
Jet transport	1,1	0,09

Then, in order to design the surfaces in detail, the same approach used for wing design can be followed. For guidelines, the reader can make reference to the Chapter 5. The only additional recommendation is to think that depending on the integration strategy of the empennages with respect to the rest of the aircraft, the effectiveness of the empennages and of the mobile surfaces may be drastically reduced. Moreover, the effectiveness of the empennages and related control surfaces may be also affected by the vehicle performances and thus, from the mission profile. Indeed, high angles of attack, high Mach numbers and high altitudes may influence the lifting capability of the empennages and for this reason, proper accommodation, should be selected. As reference, three different possible empennages layouts are reported, showing a qualitative way to understand the mutual interactions between, for example, horizontal and vertical empennages.

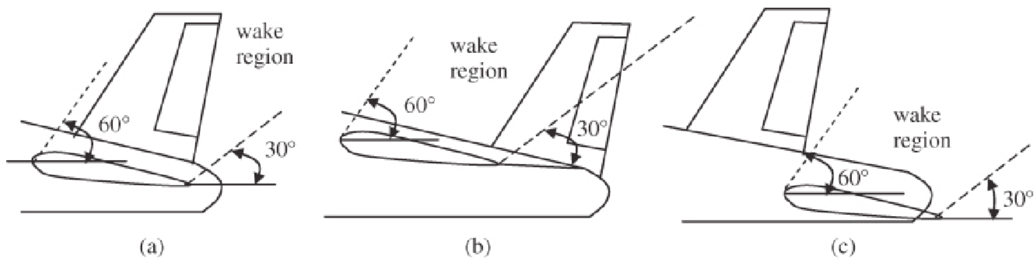


Figure 261: Tail iinteractions with horizontal empennages

In the next section, useful suggestions for the definition of the Flight control system, especially for the critical case of hypersonic vehicles are reported.

7.7.4 FCS design challenges for hypersonic vehicles

Approaching the design of a FCS for hypersonic vehicles, it has to be noticed that different challenges should be faced with. In particular:

1. It may not be possible to guaranteed the proper effectiveness of the control surfaces during the overall mission profile, especially during the most challenging phases, where high speed or high angles of attack should be performed. In these cases, it is important, since the beginning of the design process, to define whether the hypersonic vehicle can be controlled and manoeuvred exploiting flight control

surfaces or whether a different system, such as a reaction control system (RCS) may be designed and used.

2. The control surfaces may suffer from aerothermodynamic problems, due to the high temperature reached by the constituent materials. The very high temperature reached by the materials are able to affect the air that surrounds them, changing its chemical and physical properties, resulting in a reduced lifting capability. In order to reduce this problem, different strategies should be adopted since the conceptual design phase. For example, it is convenient to properly locate the control surfaces in such a way that they can be contained, during all the mission phases inside the Mach cone. In addition, proper materials should be selected in order to withstand with such temperatures and proper design suggestions should be selected. For instance, it has been demonstrated that a leading edge of the control surface, perpendicular to the main flow direction would be beneficial, allowing a more homogeneous distribution of the heat loads over the surface.
3. Sizing equations are practically derived solving the equilibrium equations written for the different flight conditions. Considering the studies and the results reported in literature, the classical equations can be exploited paying attention to the values of the several variables and of the different aerodynamic derivatives. In particular, depending on the flight altitude, different atmospheric models should be implied (Jacchia, etc) to properly evaluate the values of the variables describing the fluid in which the aircraft is operating.
4. The estimation of proper aerodynamic derivatives may be difficult. In this contest, the selection of proper reference values coming from hypersonic wind tunnel tests may be used. For instance, this has been the process followed by the ESA to size the LAPCAT MR2 flight surfaces, starting from the wind tunnels test of a small scale prototype called HEXAFLY.

7.7.5 FCS configuration alternatives

Once the tail and or canard surfaces have been designed, the selection of a proper control surfaces configuration is required in order to derive a proper Flight Control System. At conceptual design stage it is important to define the type of configuration, also looking at non-conventional ones. In the following table, the major configurations in terms of control surfaces are reported.

Table 114: Control surfaces configuration alternatives

Control Surface configuration	*Aircraft configuration*
Conventional (aileron, elevator, rudder)	Conventional Canard replacing elevator
All moving horizontal tail, rudder and aileron	Horizontal tail and elevator combined
All moving vertical tail, elevator and aileron	Vertical tail and rudder combined
Flaperon, elevator, and rudder	Flap and aileron combined
Taileron, rudder	All moving horizontal tail (aileron) and aileron combined
Elevon, rudder (or equivalent component)	Aileron and elevator combined
Ruddervator, aileron	V-tail
Drag-rudder, elevator and aileron	No vertical tail
Canardvator, aileron	Elevator as part of the canard, plus aileron
Four control surfaces	Cross tail configuration
Aileron, elevator and split rudder	No vertical tail.
Spoileron, elevator and rudder	Spoiler and aileron combined

7.8 Safety and reliability assessment for the integrated configuration

Besides the present work deals with conceptual design phase, since the beginning of the document, the crucial role of safety has been stressed. In particular, it became fundamental during the phase of integration and this is the main reason for the presence of this section in this Chapter. Moreover, the presented workflow has been set up in order to be fully integrated within the already envisaged and presented complex and multidisciplinary design methodology because it is mainly based on a SE approach. This also allows integrating the overall process within the MBSE tool-chain.

Considering the most used approaches for safety and reliability assessment, those applicable at conceptual and preliminary design level have a solid statistical base. Of course, within the scientific community dealing with innovative transportation systems, the problem of lack of statistical population to extrapolate data to be used at the very beginning of the design process, is very well-known. For this reason, the authors are firmly convinced that, due to the high-level of integration and complexity and huge costs of these future transportation systems, the only way to adopt a reasonable risk mitigation approach and to widen the public consensus is the development of a proper methodology to overcome the impossibility of direct application of semi-empirical estimations based on statistical data. Moreover, taking into account the high level of integration of the different steps of the modern conceptual design approaches, the methodology has been conceived to be fully integrated within a multidisciplinary process. Figure 262 summarizes in a flow-chart form the main steps of the proposed methodology, highlighting the qualitative-quantitative approach. In particular, it suggests a step-by-step approach that would lead to the generation and actualization of the coefficients of the semi-empirical estimations to be furtherly used in conceptual design activities of future hypersonic transportation systems.

The study of RAMS characteristics of a system since the very beginning of the design process is currently considered an unavoidable activity. Indeed, the requirements coming out from these preliminary analyses become the guide for the following steps and crucial Figures of Merit in design trade-offs.

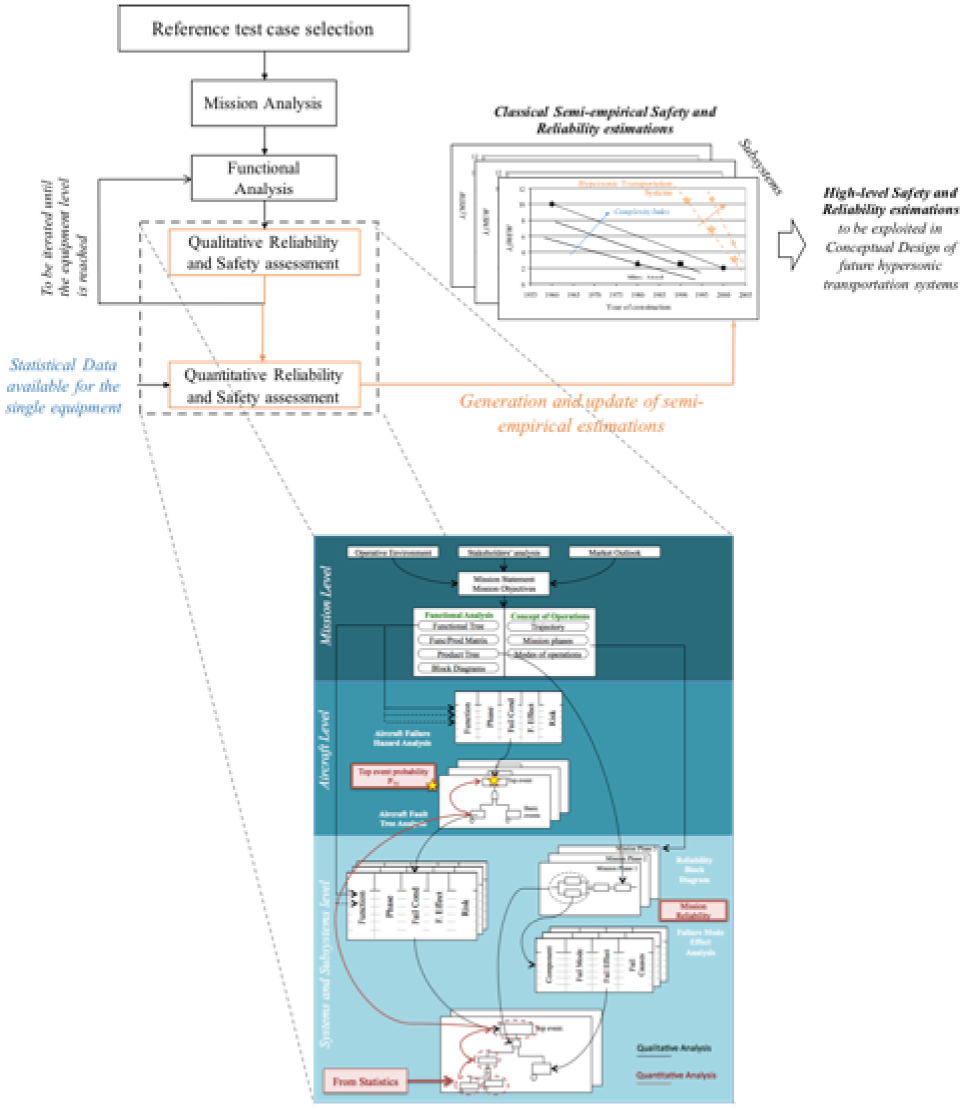


Figure 262: Integrated safety and reliability assessment methodology

Referring to the methodology proposed some years ago (Musgrave, 2009), it is possible to evaluate RAMS characteristics of a pretty new aerospace product, during the conceptual design phase, where detailed information at system or subsystem level are not yet available. For this reason, the proposed methodology uses as input parameters that are already known or of whom it is possible to make estimations. Among the widely used parameters, it is worth noticing the weight, role (type of mission), level of complexity, the date of construction, etc. Besides the level of approximation that is characteristic of these semi-empirical formulations, the highest problems are related to the lack of statistical information. But in this specific case, the population selected for the generation of the mathematical formulation results sufficient to achieve good results.

7.8.1 Reference methodology for the Safety and Reliability assessment in conceptual design

The importance of properly estimating the reliability characteristics and, especially, the logistic reliability failure rate, λ, since the conceptual design phase have already been discussed. The traditional approach consisting in evaluating the RAMS characteristics of a system only during the development phases was usually based on the assumption of an already well-defined system architecture and identified components. In this case, in order to estimate the failure rate of the overall system, it is sufficient to sum all the failure rates relative to each single component. In case the obtained estimation will not fulfil the high level requirements, often imposed by regulations (unfortunately, they are not so strict yet in the hypersonic transportation domain), it could be necessary to select components with a higher reliability or modify the architecture inserting a different number of components and redundancies. This bottom-up approach is quite difficult to be applied during the conceptual design phase and for this reason, a top-down approach is becoming more familiar in the conceptual design activities.

In order to carry-out this top-down approach, first of all it is necessary to estimate the failure rate of the overall system on the base of the existing regulations or exploiting the small amount of data available in conceptual design and then, a proper allocation of this value on the subsystems can be pursued.

Once the coefficients have been defined, the failure rate can be estimated as follows:

$$\lambda = \left(\frac{\lambda}{MEW}\right)_{MCA} \cdot IR \cdot IC \cdot IA \cdot MEW$$

where

IR is the role index

IC is the index taking into account the complexity of the system

IA is the index taking into account the historical period in which the aircraft has been developed

MEW is the Manufacturer Empty Weight [t]

$\left(\frac{\lambda}{MEW}\right)_{MCA}$ is the reference ratio statistically obtained considering Medium Civil Aircraft (1,8 failures/1000 FH/time)

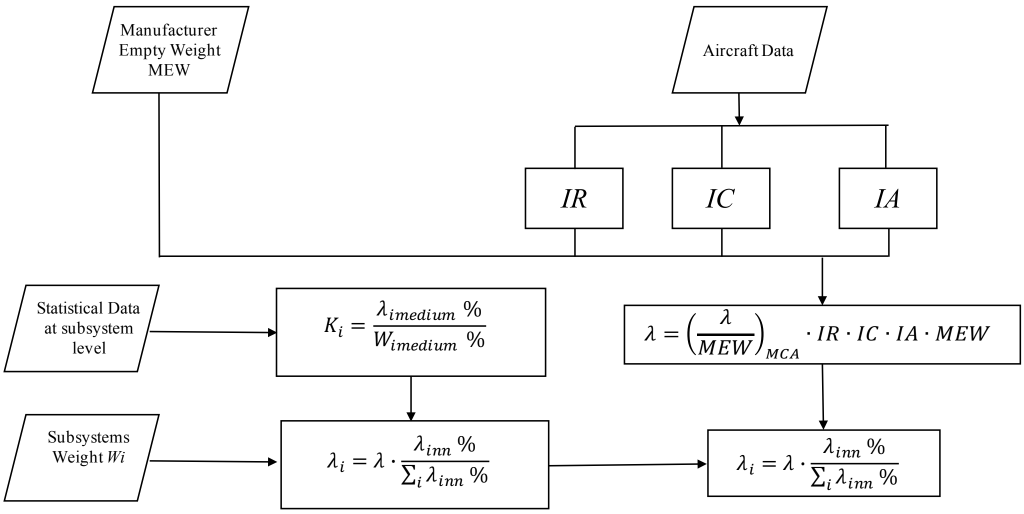


Figure 263: Flow chart for the failure rates estimation

At this point, it is possible to continue the process with the allocation of the system level value to the various subsystems mainly considering the weight parameter

As it is summarized in Figure 263, there is the need of providing an update of at least these coefficients (IA, IC, IR) in order to obtain useful data for realistic estimation of the RAMS characteristics in conceptual design.

7.8.2 Methodology for the update of the semi-empirical safety and reliability estimations

As it was announced above, the here proposed methodology consists of two major parts:

- A Qualitative Analysis where, starting from the top-level design activities, a safety assessment is performed following a top-down approach, from the mission level to the equipment level. This process does not imply any quantitative evaluations
- A Quantitative Analysis where, starting from the results of the Qualitative Analysis and exploiting available statistics at equipment level and following a bottom-up approach, it is possible to retrace the way to derive the probability of the top-event related to the mission or to the system.

The following paragraphs aim at describing in detail the two parts of the methodologies, highlighting the fact that the proposed approach is fully integrated within a modern conceptual design activity based on a Systems Engineering (SE) approach.

7.8.2.1 Qualitative Process

This subsection aims at providing a detailed description of the sequence of actions summarized inFigure 264. This first part of the methodology aims at deriving the basic elements composing the high-level system, starting from the identification of all the possible people and public or private entities interested in the design of the innovative aerospace product and in all the possible advantages coming out from its exploitation.

It should be noticed that the very first step, i.e. the Stakeholder analysis, coincides with the usual starting activity of each conceptual design of innovative products, consisting in looking for all the possible interested private or public entities and trying to evaluate their needs and desires. At the same time, as soon as the product category has been defined, it is important to deeply investigate the market in order to understand where to sell the product and also to evaluate if the technologies that should be developed for this application can be interesting in different fields. Moreover, a review of the national and international regulations related to the development but also to the operations of the vehicle should be considered and a first draft list of constraint can arise.

From this preliminary analysis, a mission statement and consequently a first list of objectives and requirements could be derived. At this point it is possible to concretely starts the safety assessment creating a simple Functional Tree. Within the conceptual design phase, with a Systems Engineering approach, Functional Trees are used to look at the system from a broader functional perspective, allowing to derive all the functionalities or capabilities the System should guarantee to fulfil the main objectives. While these functions are associated to subsystems, equipment and components able to perform these functions, they could also be exploited as inputs to carry out the Functional Hazard Assessment (FHA) at aircraft level.

Functional Hazard Assessment (Figure 266) is a logical examination of functions to identify and classify failure conditions related to those functions according to their severity (SAE, 1996). The objective of the FHA is to consider functions at the most appropriate level and to identify failure conditions and the associated classifications while considering both loss of functionalities and malfunctions. It is important to notice that the FHA, especially if carried out at system level or at a lower one, should identify the failure conditions for each phase of flight when the failure effects and classifications vary from one flight phase to another. The FHA also allows to derive safety requirements needed to limit the function failure effects which affect the failure condition classification. Once the high level requirements have been identified, they may be used to generate lower level requirements. As well as for all the other categories of requirements, this process shall continue iteratively, until the design process is complete. The most common and useful way to perform a FHA is to create a table view where this data can be organized, as part of the Preliminary System Safety Assessment process for the systems or items.

It is worth to noticing that in Figure 264, where the overall process is shown, there are two levels of FHA, the Aircraft level FHA and the System-Subsystem level FHA. Indeed, the FHA will be carried out at different levels of the design process but exploiting the same principles. Coming back to Aircraft level, once the functional tree and FHA have been derived, each failure condition identified by the FHA should become the top-event of a Fault Tree.

Fault Tree Analysis is a deductive, failure-based approach that starts with an undesired event (called top event) and then logically determines (deduces) its causes using a systematic, top-down approach. In determining the causes, a Fault Tree (FT) (Figure 265) is constructed as a logical illustration of the events and their relationships that are necessary and sufficient to result in the top event. To carry out FTA, a real tree consisting in boxes and connectors should be built. In particular, the types of boxes should be used to identify the different kind of events, while the different connectors stand for the Boolean algebraic symbols ("AND" and "OR") and should be used to specify the relationships among the several events. The FT is just a qualitative model, but it provides extremely useful information on the causes of the undesired event. The FT can also be quantified to provide useful information on the probability of the top event occurring and the importance of all the causes and events modelled in the FT. In this way, a FTA can be carried out taking into account that each basic event of all the FTs will become the new failure condition for a specific function of a lower level FHA. Moreover, following the procedure explained for the Aircraft level FHA, this lower-level FHA should also receive inputs from Functional Tree carried out for the relative design level.

Then, to continue in the analysis, it is necessary to move from a strict functional view of the system to a more product-based stand point. This is usually performed within the design procedure based on SE approach, linking the results of product trees with the possible way of working of the system itself or its behavior during its operative life, creating the so called Concept of Operations (ConOps). The Concept of Operations allows describing how the system will be operated during its entire life cycle, in order to achieve the mission objectives. Typical analyses contained in ConOps include evaluations of mission phases, operation timelines, operational scenarios, end-to-end communications strategy, command and data architecture, operational facilities, integrated logistic support and critical events.

Carefully evaluating the results of the Concept of Operations analysis and taking into account the results of the functional analysis, the Reliability Block Diagrams (RBDs) could be derived. Reliability Block Diagrams (Figure 265) are graphical representation used to reproduce the way of working of a certain system or subsystem in a well-defined mission phase and operative mode. Indeed, depending on the operative modes, the system could be schematically represented through different layouts. Exploiting existing reliability theories, it is possible to translate the scheme in an algebraic equation in which the known values are the failure rates of the different components and the unknown parameter is the system or subsystem Reliability. Then, on the one hand, each component of the RBD can be in depth evaluated from the safety stand point exploiting the Failure Modes and Effects Analysis (FMEA), while on the other hand, the way in which the different components are mutually interfaced will define the logic operators of the related Fault Tree. The Failure Modes and Effects Analysis is a systematic analysis of the way in which each subsystem or component can be affected by malfunctions, thus behaving differently if compared to what it was expected in nominal mode.

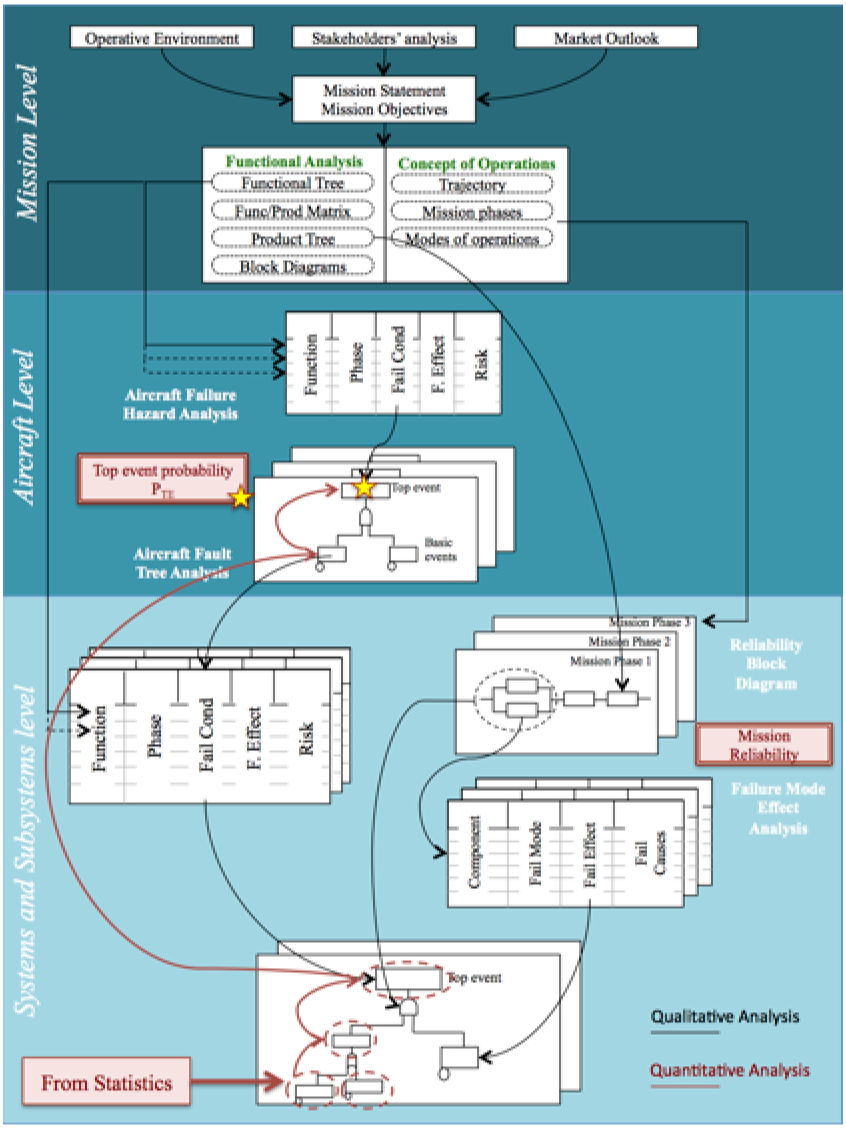


Figure 264: Flow-chart summarizing the qualitative phase

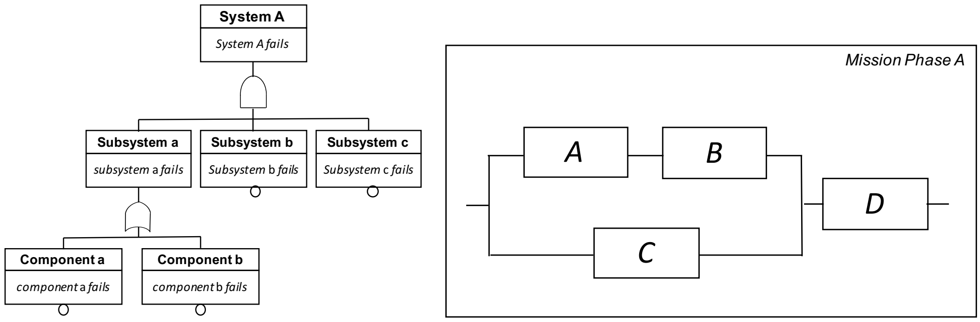


Figure 265: Fault Tree (left) and Reliability Block Diagram (right)

Function	Phase	Failure Condition	Failure Effect	Risk

Component	Failure Mode	Failure Effect	Failure Causes

Figure 266: Failure Hazard Analysis (top) and Failure Mode and Effect Analysis (down)

For each type of failure, this analysis allows to induce the effect related to this failure that could be experienced by the system. Then, starting from the failure modes and from the possible causes of these mishaps, the failure effects and its seriousness can be estimated. Similar to what has been presented before, also in this case, it is possible to exploit this tool in an iterative way in order to obtain at each step, a new set of more detailed information.

At the lowest level of decomposition (equipment level) the failure effects of the FMEA are the basic events of the FTA.

Eventually, this first part of the methodology will be used to derive the functional and behavioural structure of the system, reaching a decomposition at a sufficiently low level such that it is possible to assign numerical values of failure rates to each basic component. This will be the first activity of the quantitative methodology (bottom-up approach), in-depth analysed in subsection “Quantitative process” that would lead to the estimation of an aircraft level failure rate to be compared with existing regulation and/or high level constraints.

7.8.2.2 Quantitative Process

Once the activities described in the previous section have been completed, the quantitative analysis can start. The process is summarized in the same Figure 264 where the process followed should be read from bottom to the upper part. At the beginning, it is necessary to consider the lowest event that could occur (i.e. the event related to a malfunctioning of one of the lowest-level identified component) and the smallest identified components of the system and associate the probability of occurrences to each of them. It is important to notice that at the beginning of this process, a deep analysis and research of available statistics shall be conducted. Then, it is reasonable to proceed with a bottom-up approach aimed at solving the aforementioned probability equations, reaching the top-level event as graphically summarized in Figure 264. In particular, the quantitative process exploits all the Fault Trees previously derived in the qualitative process, starting from the lowest level until reaching the aircraft-level Fault Tree.

Exploiting a similar approach, in addition to the Top Event Probability, estimations of Mission Reliability can also be carried out, solving the Reliability Block Diagrams derived for each mission phase.

7.8.2.3 Application to the selected reference case study

This subsection describes the result of the application of the reliability and safety assessment methodology described above in order to integrate the air-breathing propulsion subsystem in such a way that it would be able to fulfill all the set of mission requirements with an acceptable risk level. As far as the air-breathing is concerned, the envisaged subsystem is composed of two main engines each of which should be able to guarantee the thrust required to overcome the maximum take-off weight enabling the VTOL capability (Figure 267). Due to the shape of the vehicle in which they should be accommodated and the presence of the other subsystems imposes the two engines to be placed on the two sides of the rocket motor. They are equipped with two main steerable nozzles but there is also the possibility of conveying the hot gasses in a distribution system to feed the four secondary steerable and retractable nozzles installed in the lower flat surface of the vehicle. Two main fuel tanks located in the wing available room, through a cross-feed valve can feed both the two engines. At the same time, for safety reasons, another cross-valve is required in the hot gasses distribution lines in order to allow the left-hand engine to provide hot-gasses to the right-hand placed nozzles and vice versa, in case of One Engine Inoperative (OEI) condition. In addition, component like pumps or valves should be obviously taken into account for the safety assessment.

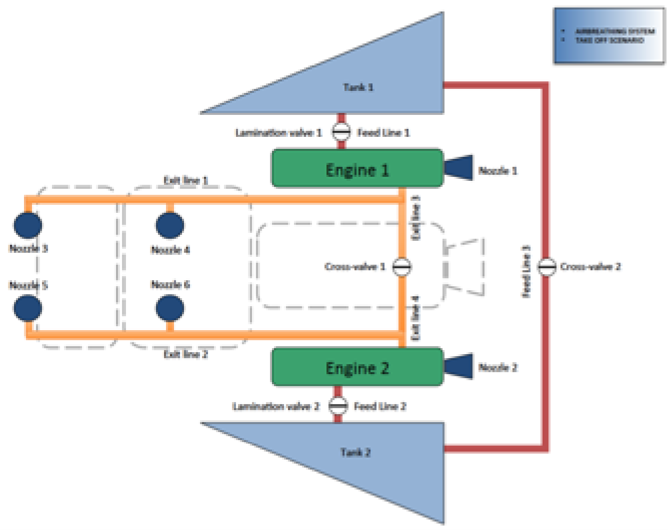


Figure 267: Air-breathing subsystem for the reference case study

The approach proposed above has been applied to identify the top-event probability of the so complex propulsion system able to guarantee VTOL capabilities. Some example of the Fault Tree and Reliability block diagrams are reported in Figure 8 and 9. It is clear that for each foreseen operative mode of the propulsion system, a different RBD and different FTAs have been sketched. Here is only an example of these diagrams derived for the most critical phases of the mission: the take-off. As it is easily noticeable, the analysis should be carried out for each single steerable nozzle placed in the bottom of the vehicle but no information is required for the two main nozzles, and this is due to the hypothesis that the overall exhaust mass flow is diverted in the bottom placed ones. Moreover, additional complexity of the scheme is added by the fact that different kinds of cross-feed have been envisaged in order to enhance the level of safety.

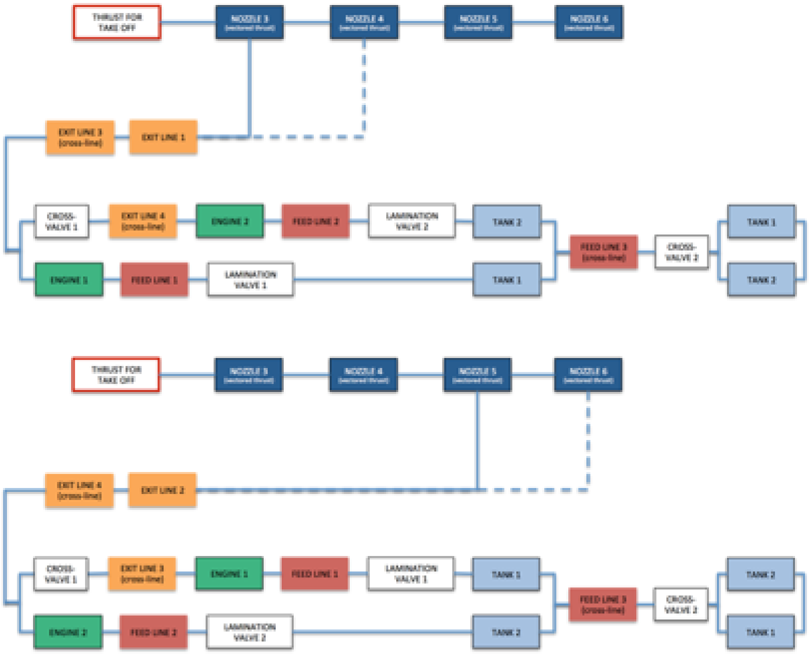


Figure 268: Example of RBD for the reference case study

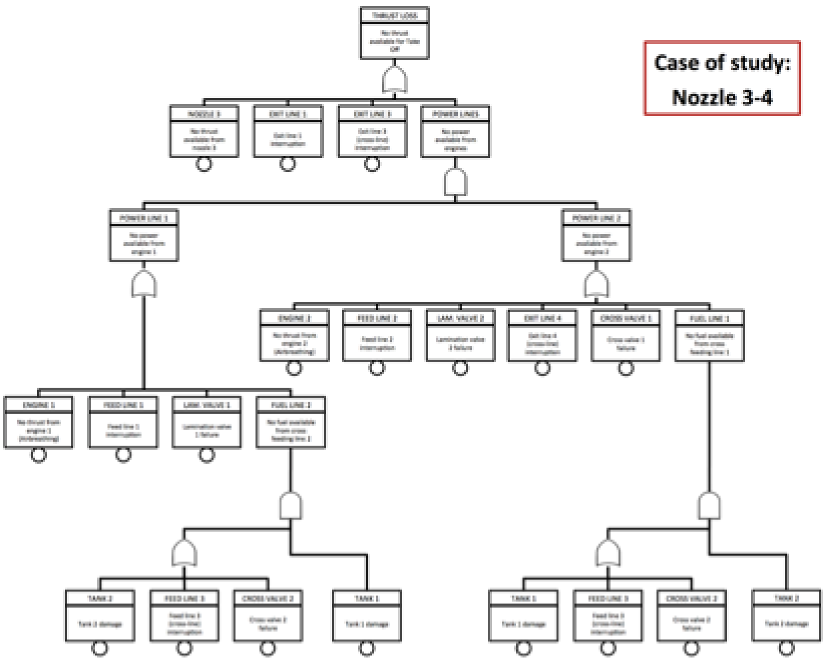


Figure 269: Example of FT for the reference case study

As it has been explained in the methodology part, each tree can be summarized in an equation that can be solved starting from the failure rate of the single components. Data coming from statistics have been selected in Military database. This could be one of the main reasons for which the expected propulsive system failure rate (equal to 0.9118 10^{-2} failures/1000h) is closer to a fighter than to a civil aircraft (Table 115). Of course, this result is acceptable because of the high complexity of the system. In particular, Table 115 summarizes the results obtained from the application of the methodology to the case-study presented in the previous subsection. The estimated number of failures/1000 FH puts the suborbital vehicles between fighters and military transportation systems. This first result is in line with the expectation considering the high level of complexity of the considered subsystems and the existing reference on-board systems architectures selected. However, in perspective, it would be useful to improve the design of the subsystems, acting on redundancies or basic components selection in order to enhance the reliability level and obtained estimations closer to civil transportation. This would ease the future certification process of such vehicles.

The application of the methodology proposed in this paper to some case studies allows a first preliminary attempt of update of the semi empirical models. In particular, the tables reported in this section show the estimated values for the specific case of the suborbital vehicles, as preliminary approach to the field of hypersonic transportation systems.

In particular, looking at the estimation of the Role Index (Table 116), both suborbital vehicles and point-to-point transportation system has values comparable with military transportation system. In this case, no distinction within the family hypersonic transportation systems seems to be necessary because, besides the wide range of Mach numbers and cruise altitudes, the Index Role is mainly affected by the type of mission profile.

Conversely, a different categorization has been proposed in order to estimate Complexity Indexes (Table 117). In particular, considering that the take-off strategy can deeply affect the complexity of the vehicle and related missions, for the suborbital vehicles, two different level of complexity have been estimated, differentiating between VTOL and HOTOL vehicles. On the other hand, considering that the level of complexity if a point-to-point hypersonic transportation system is strictly related to the technology readiness level associated to the selected components, two levels of complexity have been estimated depending on the use of state-of-the-art or advanced technologies. The same classification has been considered in order to estimate the technology aging coefficient (Table 118). In particular, considering that currently under-development hypersonic transportation systems will be developed in different time frameworks.

Table 115: Failure rates (Chiesa, 1990)

Role	***Propulsive System failure rate [failures/1000 FH]***
Fighter	$1.809\ 10^{-2}$
Military transportation	$0,226\ 10^{-2}$
Civil transportation	$0.00263\ 10^{-2}$
Suborbital vehicle	***$0.9118\ 10^{-2}$***

Table 116: Role Index update

Role	***Role Index, IR***
Fighter	16,6
Military transportation	2,1
Civil transportation	1,0
Suborbital vehicle	2,1
Point to point vehicle	2,1

Table 117: Complexity Index update

Complexity Level	***Complexity Index, IC***	***Reference Vehicles***
Very Low	0,5	Ultra-light aircraft
	0,6	FIAT G91
Low	0,8	S211
	0,9	ATR 42
Medium	1,0	AMX, G222
High	1,4	TORNADO, EF2000, F35 CTOL, A320, ***HOTOL Sub-orbital aircraft*** ***Hypersonic civil transportation aircraft with state-of-the-art technologies***
Very High	1,6	F22, F35 VTOL, ***VTOL Sub-orbital aircraft*** ***Hypersonic civil transportation aircraft with innovative technologies***

Table 118: Technology Aging Index update

Development Year	*Technology aging Index, IA*	*Reference Vehicles*
1950s	3	FIAT G91, F86
1960s	2,5	F104S, Caravelle
1970s	2,0	TORNADO, DC9
1980s	1,5	AMX, ATR 42
1990s – 2000	1	EF2000, A320
2010-2020	0,9-0,6	F22, F35, ***Sub-orbital aircraft*** ***Hypersonic civil transportation aircraft with state-of-the-art technologies***
2030-2040	0,5	***Sub-orbital aircraft*** ***Hypersonic civil transportation aircraft with innovative technologies***

Eventually, considering the author intends to fully implement the approach within the proposed Model Based Systems Engineering tool-chain, enlarging it to specific safety and risk assessment tools that will also allow to trace the safety requirements all along the conceptual design activity. Furthermore, aiming at increasing the level of confidence of these indexes, other case studies have been identified and there are currently some under-development activities (e..g. LAPCAT MR2 and SpaceLiner). In particular, future application of the methodology will be supported by HyDat, the under-development database of hypersonic programmes, projects and activities.

7.9 Integration of a Cabin Escape System

7.9.1 The need of a Cabin Escape System for suborbital vehicles aimed at parabolic flights

Considering space manned missions, it is quite immediate to think about systems able to guarantee crew and flight participants survivability. But why, for aeronautical missions, this is quite taken for granted? In order to understand the major reasons why astronaut crews should have an escape option and airliners passenger do not, some safety numerical evaluations have been performed in the past, considering the Risk of Loss of Life during flight as Figure of Merits (Musgrave, 2009). In particular, Table 119 summarizes the results of the Safety and Mission Assurance Directorate of NASA. Despite of the high number of initiatives all around the world, considering the experimental/demonstration level of this vehicles, it is convenient to currently assume as reference value, the estimation characterizing the current Human Spaceflight initiatives (last row of Table 119). However, it is important to aim at reaching a higher level of safety (pointing towards commercial aviation as a target) in order to enhance the public consensus.

Looking at the differences in values reported in Table 119, it could be interesting to understand which could be the major causes and think about corrective actions to be taken in the conceptual design activities to keep closer to the target. Considering the main factors for which suborbital flight are still too risky, it could be notice that the major role is played first of all by the low System Readiness Level. This parameter depends not only on the TRL (Technology Readiness Level) but also on the way in which the different components are integrated within the system. In addition, the limited number of flight hours with respect to the other categories, the very hazardous environment in which the vehicle is operated and the restricted possibilities of carrying out flight tests are the major causes of the high level of risk. Unfortunately, the need for developing ad-hoc systems to be used only in case of emergency represents an unavoidable increment in complexity and weight, and, in the past, it has been avoided preferring additional levels of redundancy. Since the beginning of the design process for the reference case study, trade off analysis between risks and costs has been performed in order to select the optimal strategy for guaranteeing crew and passengers survivability. Considering the specific case of suborbital flights, taking into account the short duration of the mission, the limited number of passengers and the need to accommodate non-trained people in a high comfort level environment, after the evaluation of other safety solutions, like ejectable seats or escaping pods, cabin escape system has been selected as the optimal solutions.

Table 119: Risk of Loss of Life for different kind of aerial transportation systems.

Type of flight	Risk of Loss of Life During Flight
Commercial Airplane	$1/10^6$ flight hours
Military Aircraft	$1/10^5$ flight hours
Combat in a military jet Aircraft	$1/10^4$ flight hours
Human Spaceflight	$1/10^2$ flight hours

Under the hypothesis of a systems consisting of a detachable cabin, the probability of crew and passengers survival is enhanced like demonstrated with the following formula.

$$P_{crew\ suvival} = 1 - (P_{primary\ failure})(P_{rescue\ failure})$$

where:

$P_{crew\ suvival}$ is the probability of survivability for the crew and passengers;

$P_{primary\ failure}$ is the probability for the vehicle to experience a failure requiring a crew escape system.

$P_{rescue\ failure}$ is the probability that the rescue system and related procedures fail in their mission.

This is the general statistical formula that could be adopted to evaluate the crew survival but it is important to notice that, depending on the safety approach, it can slightly vary. For example, in case of future transportation systems able to carry a high number of passengers and thus requiring more than a flight to complete the rescuing, or more rescue systems operated in parallel, the formula can be modified as follows:

$$P_{crew\ survival} = 1 - \left(1 - P_{primary\ success}\right)\left[1 - (P_{rescue\ success})^{n\ requested\ flight}\right]$$

where:

$P_{primary\ success}$ is the probability for the vehicle to complete a mission successfully.

$P_{rescue\ success}$ is the probability that a rescue activity is completed positively.

For the sake of clarity, it is worth noticing that the probability of crew survival is strictly related to the mission phase in which the catastrophic event may happen. Indeed, the possibility of actuating the escape system it depends on the altitude and speed regime of each mission phases. This has been also clearly outlined by several post-processing analyses of the Space Shuttle major incidents.

7.9.2 Integration of the Cabin Escape System

The design and sizing of a Cabin Escape System (CES) has to be considered from a holistic point of view and should be taken into account since the beginning of the design process, because it can deeply impact on the design of the entire vehicle and, especially, to its architecture. At first, it is necessary to carefully decide the location of the cockpit and passengers compartment to guarantee the possibility of detaching it from the main vehicle but it is also crucial to determine which subsystems should be installed within the detachable parts and which ones could be hosted outside, in the main part of the vehicle. The presence of a CES deeply affects the design of the entire transportation system, especially in a particular case-study with the constraint of the vertical take-off and landing strategy and of a single stage. In particular, envisaging a CES implies additional constraints for the selection of the proper location of the cockpit and passengers compartment within the architecture of the transportation system. Among the several issues impacting on this selection process, the following aspects should be taken into proper consideration:

- ***Aerodynamics***: the CES, once detached from the main spacecraft, should be able to safely transport the passengers on ground, thus, a proper aerodynamic behavior should be guaranteed. In particular, the design of a CES with a proper lift-to-drag ratio can allow to perform a controlled re-entry, minimizing the structural and heat loads experienced by the CES.
- ***Accessibility*** for crew and passengers: considering an emergency condition, the CES should be able to allow passenger egress during whichever type of landings, both on ground and over-sea, in floating conditions.
- ***Maintainability***: considering the need of guaranteeing fast turn-around time in order to comply with the need of the stakeholders requiring a routine service, the CES should require a limited number of maintenance actions. In particular, the safety-critical components (like the separation mechanisms) with a major expected frequency of scheduled maintenance, should be easily accessible to the maintainers. In particular, concepts like in-Line Replaceable Units (LRU) should be envisaged.
- ***Safety:*** Besides the CES aims at enhancing the safety levels of the flight participants, this system should not increase the level of risk of the on-ground personnel and of the overflown populated areas.

Considering all these aspects, the need of developing a detachable compartment and the safety requirements forced the designers to it as far as possible from the main propulsion subsystem, consisting of the two air-breathing engines, with related fuel subsystem, and the rocket, with the related propellant subsystems (Figure 270, Figure 271). Indeed, the presence of a rocket engine increases the risk of hazardous events mainly due to the degree of explosiveness of the propellants stored on-board.

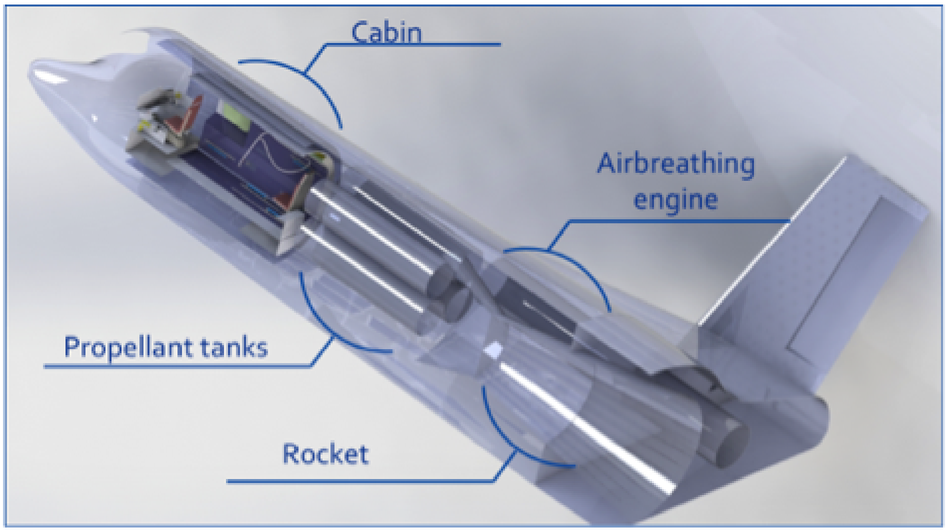


Figure 270: Major subsystems integration within the entire transportation system

Figure 271: Pictorial view of the CES concept.

From the subsystems integration standpoint, it is important to understand which functionalities should be guaranteed to the CES when detached and as consequence, the main equipment that should be installed. Then, in a second step, it is important to verify if it could be possible to install the system within the available volume of CES or to locate it in the main part of the vehicle, considering a redundancy in the passengers' compartment. In order to understand which subsystems should be considered, functional analysis has been carried out in order to derive the major functionalities to be guaranteed to the CES. Following the methodology proposed in past above-mentioned works of the research group, and applied in different context within the aeronautical and space domain, the results proposed in Figure 270 and Figure 271 have been obtained.

It is clear that in order to enhance the safety level of passengers (without increasing the risk of inhabitants of populated areas overflown during the mission), the design of the vehicle shall take into account the following list of requirements. To Be Defined (TBD) values will be explicated thanks to future analyses.

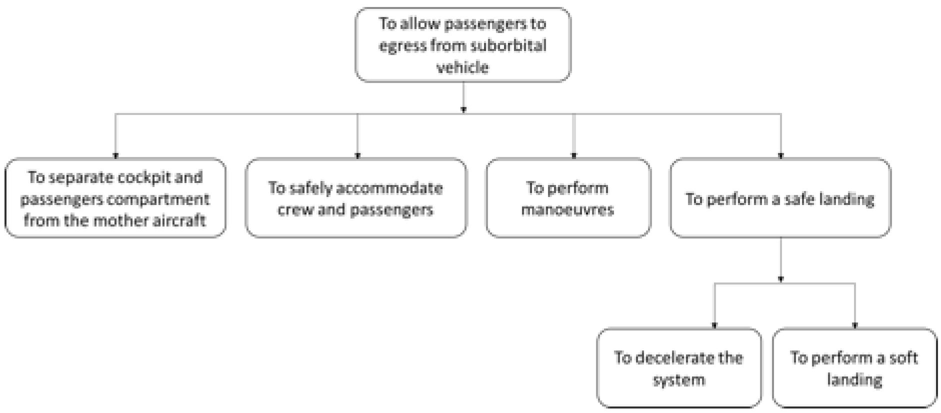


Figure 272: Functional Tree for the CES concept

	Subsystems						
		Cabin	Flight Control subsystems	Parachute	Landing bags	Separation mechanisms	Guidance Navigation and Control Reaction Control System
Functions	To separate cockpit and passengers compartment from the mother aircraft						
	To safely accommodate crew and passengers						
	To perform manoeuvres						
	To decelerate the system						
	To perform a soft landing						

Figure 273: Function/Product marix for the CES concept

Requirements mainly impacting on the structure and internal layout of the CES

- The CES shall be able to safely accommodate the crew and 4 passengers.
- The CES shall be able to safely accommodate the additional payload consisting in scientific experiments.
- The CES shall enhance the safety levels during all the mission phases.
- The CES shall be able to support the maximum aero-structural and thermal loads.
- The CES shall guarantee fast ingress and egress of people.
- The CES shall be able to guarantee a TBD lift-to-drag (L/D) ratio during all the mission phases.
- Structural integrity of the detached configuration should be guaranteed during the different mission phases
- Proper high temperature resistant materials should be exploited.

Requirements mainly impacting on the separation subsystem of the CES

- The CES shall be provided with a fast and reliable separation subsystem
- During the separation, flight participants shall not experience g-loads over TBD m/s^2.
- During the separation, mobile surfaces should be detached, properly cutting out the structure of the bottom flat part of the fuselage.
- The CES shall be able to leave the main vehicle in TBD seconds, avoiding to be involved in possible explosion of the main part of the vehicle

Requirements mainly impacting on the propulsion subsystem of the CES

- The propulsion subsystem shall allow the CES to be exploited during the overall mission profile.
- The propulsion subsystem shall allow the CES to separate within the safety margins.
- The propulsion subsystem shall enhance the CES controllability when detached.
- The propulsion subsystem shall minimize the additional weight for the overall configuration.
- The propulsion subsystem and related propellant subsystems shall minimize the additional risk of explosion.

Requirements mainly impacting on the control subsystem of the CES

- The CES should be manoeuvrable within the lower atmospheric layers.
- The CES should be controllable within the outer atmospheric layers.

Requirements mainly impacting on the design of deceleration and landing devices for the CES.

- The CES should be equipped with a proper deceleration subsystem to be used in the outer atmospheric layers.
- The CES should be equipped with a proper deceleration subsystem to be used in the inner atmospheric layers.
- Proper landing devices should be adopted to guarantee crew and passengers survivability in case of impact with ground.
- Proper landing devices should be adopted to guarantee crew and passengers survivability in case of impact with water surfaces.

Requirements mainly impacting on the environmental control and life support subsystem (ECLSS) for the CES.

- A proper ECLSS shall be envisaged for the CES.
- The ECLSS shall be able to control pressure, temperature and humidity also when CES is detached.
- The ECLSS shall be able to remove CO_2 gases and monitor the atmosphere's composition also in when CES is detached.
- Redundant oxygen tanks and related handling subsystem shall be properly envisaged to be used in case of failures of the primary ECLSS.
- ECLSS shall be capable of guaranteeing TBD minutes of operation.

Additional Requirements

- Emergency locator transmitter shall be installed within CES
- Flight Data Recorder and Cockpit Voice Recorder shall be installed within CES
- Proper avionics subsystem shall be installed within CES to allow all the guidance, control and communication activities to carry out the emergency mission.

- Proper antenna and radar shall be installed within the CES and should be able to operate also when CES is detached.

Then, the main required subsystems can be properly designed and sized with the aim of fulfilling the major requirements elicited from the mission-level functional analysis. As it has been anticipated above on, many driving factors forced the designers to locate the front fuselage.

In order to have the possibility of properly detaching the CES as sketched in Figure 271, the structure should be properly designed and sized to survive to the loading conditions envisaged for the mission. In particular, from the structural point of view, some main areas could be envisaged:

- Left small wing surface
- Right small wing surface
- Crew and passenger compartment
- Available Room for subsystems

7.9.2.1 Cockpit and passengers compartment design

The cockpit and the passengers compartment is the core of the cabin escape system. Cylindrical shape that avoids structural complexities and maximizes the available volume has been selected for the pressurized part of the fuselage (Figure 275). The minimum room to host the passengers, which guarantees them to enjoy floating in microgravity, has been accurately evaluated taking into account both aeronautical and space regulations. Moreover, different seats configurations have been evaluated to select the best compromise between comfort during flight, encumbrance and the capability of been stored or at least reduced in volume during the microgravity experience. Remembering that one of the stakeholders' expectations was to guarantee an amazing view of the Earth, the absence of windows could appear strange (a part from the security hatch). Indeed, considering the demanding mission profile, the external structural loads and the need of avoiding additional weight, it was decided to provide the external visibility exploiting virtual reality subsystem, consisting of a series of externally mounted cameras and O-LED panels covering the overall internal surface. In this way, the passengers can feel like flying freely in the sky. Psychological effects of this kind of systems have already been evaluated.

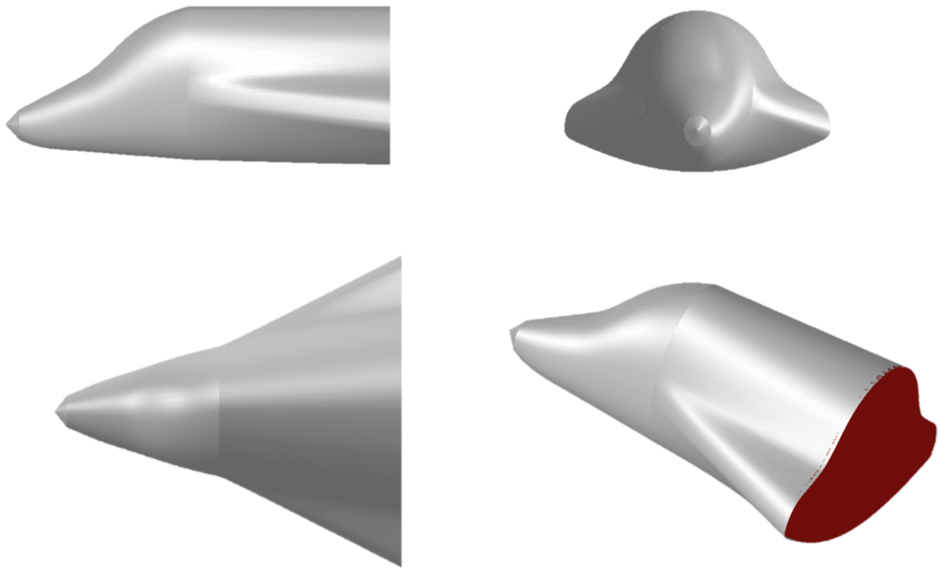


Figure 274: CES 3-views

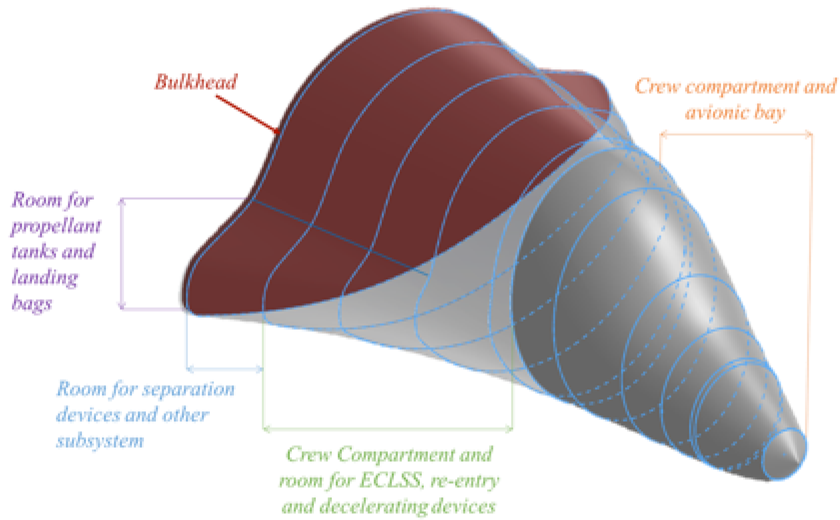


Figure 275: Details of the CES structure and volume allocation.

The thickness of the different structural elements will be fully investigated during the preliminary design phase and, to this purpose, a special role will be played by mission simulation. Indeed, the identification and simulation of the re-entry trajectory of this object in the worst scenario, will provide the designers with the major data about structural and thermal loads during the trajectory. These analyses will provide useful information for the design of the hot structures and the thickness of the ablative materials that will eventually cover the lower surfaces of the vehicle and, of course, of the cabin escape system. From the geometrical point of view, the overall sizing of CES is reported in Table 120.

Moreover, considering the previous and current studies related to the overall vehicle design and exploiting the features of CAD, after the designation of the proper materials to be used, a detailed mass estimation of the structure and of the integrated subsystems, an overall mass of the CES will be obtained. Moreover, additional stability assessment could be carried out.

Considering the current status of the research activities, the overall geometrical characteristics have been estimated and are summarized in Table 120.

Table 120: CES sizing results

Geometrical Feature	**Value**
Overall Length (rocket nozzle and mobile surfaces excluded)	8.4 m
Overall Width	6.4 m
Overall Height	3.7 m

7.9.2.2 Separation Mechanism

Considering the overall mission concept, the separation can be considered the riskiest and complex phase. Indeed, in this moment, the CES shall be able to safely and fast separate from the main transportation system. For this reason, the separation mechanism shall be installed behind the passengers’ compartment. In particular, for this kind of application, mechanically initiated pyrotechnic devices have been used. In space applications, a mechanically initiated explosive device could be very simple equipment, activated by a spring loaded firing pin that strikes a common percussion primer (Musgrave, 2009). A blow that causes metal deformation, which then pinches a small amount of a pressure sensitive pyrotechnic material between the deformation and an internal anvil, strikes this percussion primer, or cap. The sensitive powder then ignites, setting off the explosive train. A proper controller should be in depth studied in order to evaluate the firing condition, (typically an out-of-nominal set of parameters) depending on the mission phase. In this way, the escape system could be safely separated from the rest of the spacecraft in relatively short time and avoiding additional complexities.

In order to overcome the problem of pyrotechnically initiated mechanisms, alternatives have been evaluated. In particular, the non-pyrotechnic release devices developed by the Air Force Research Laboratory have been considered (Peffer, 2000). Moreover, in order to comply with the strict requirements of capsule separation from the transportation system, a small device able to generate the required thrust to guarantee the separation of the two main elements has been hypothesized.

7.9.2.3 Rocket propulsion integration

In order to guarantee a proper separation distance from the main spacecraft in case of emergency, a rocket motor should be exploited. Moreover, whether the propulsion system and the propellant tanks would be properly sized, they could be exploited to generate thrust to assist the re-entry and descent phase of the CES mission. As it is shown in Figure 276, the rocket motor should be placed just behind the passengers compartment and proper room should be envisaged in order to host the relative propellant tanks. In particular, looking at the overall escape system configuration, cylindrical tanks could be hosted in the available room within the small portion of wing (Figure 276).

7.9.2.4 FCS and RCS integration

The need of guaranteeing a complete manoeuvrability of the escape system, once it is detached together with the need of reaching specific landing site or avoiding populated impact areas, force the designers to envisage an embedded simple and light Guidance, Navigation and Control subsystem (GNC) and Reaction Control System (RCS) that could be based on existing chemical rockets or on innovative electric thrusters. In this concept, thrusters have been envisaged in order to allow attitude control during the re-entry phase. It is important to notice that in the nose of the vehicle at least four couples of thrusters should already be present to guarantee controllability of the entire configuration in nominal condition. This means that in order to be exploitable also for the control of the escape system, tanks and related subsystems should be installed in the fore part of the fuselage.

The solution proposed in Figure 276 is characterized by the presence of mobile surfaces that could be especially exploited during the latest part of the re-entry trajectory, in the lower atmospheric layers. Moreover, if properly sized, they could also be used together with the RCS subsystem to enhance the manoeuvrability of the CES at high altitudes. Considering the complexity of this solution, the Flight Control Subsystems successfully flown over IXV (Tumino, 2008) has been taken as reference. Indeed, considering that the environmental conditions experienced by IXV during its re-entry phase, whether properly scaled, could be similar to those envisaged for the emergency trajectory of the CES, proper system architecture (Victor, 2016) sizing and selection of the resisting materials could be considered as reference.

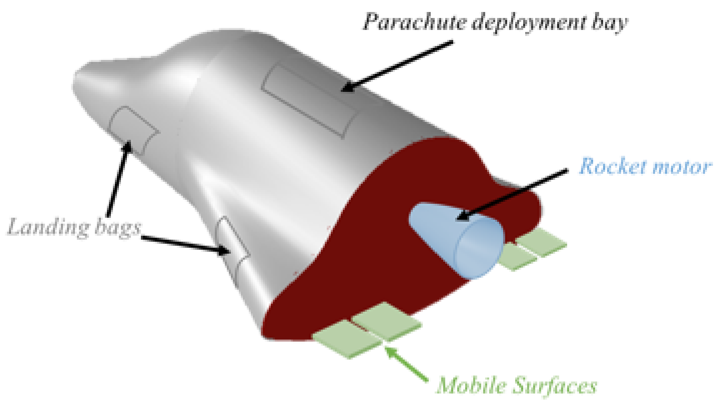


Figure 276: CAD view of the CES with focus on peculiar subsystems

7.9.2.5 Landing support

As it has been stated since the beginning of the design process, the CES should be able to safely land on-ground or over-sea, guaranteeing the survivability of the non-trained passengers that are hosted within the passengers' compartment.

First of all, this implies that proper simulation model should be built and run in order to derive the best re-entry corridor in case of emergency. In this case, the methodology suggested in (De Vita, 2015) could be exploited. It is worth to notice that this should be theoretically assessed for each point of the mission or at least for the critical point of each mission phase. Once the flight envelope of the clean configuration of the CES has been obtained, in order to guarantee the fulfilment of the maximum accelerations requirements, a proper parachute subsystem should be sized and the simulations repeated for the clean configuration plus the addition drag determined by the parachutes, in a typical iterative process, until requirements satisfaction. Please notice that the use of simulation is strictly required in order to model the complex behaviour of the parachute itself and of the entire subsystem (CES with deployed parachute). In particular, stochastic models for the prediction of the deviation of the trajectory with respect to the nominal one (clean CES configuration) could be exploited.

Then, with the aim of supporting the landing and soften the contact with ground or water surface, proper devices should be installed. In particular, considering that, at least at the beginning of the spaceplane era, only over-sea trajectory would be permitted by the safety agencies, the CES should be equipped with small inflatable bags that would guarantee a softer landing and the floating capability. In this case, it is worth to notice that the size of the bags is strictly dependent to the maximum available time that should occur to the emergency rescue boat to reach the landing place and save the crew and passengers. For this reason, a model representing the overall System of Systems, including the overall on-ground segment, should be taken into account and simulated since the conceptual design level.

7.9.2.6 Environmental Control and Life Support System integration

The Environmental Control and Life Support System is the typical subsystem that makes a distinction between manned or unmanned space systems, being aimed at satisfying crew and passengers' needs and providing resources for their activities. This system shall support the crew and passengers compartment with the proper pressurized environment, controlling temperature, pressure and air composition and avoiding any type of contaminations. Considering the currently available technologies, this system could not be entirely hosted within in the available room under the floor of the passengers compartment. However, considering that in worst scenario, in which a severe failure happens at the top of the parabola, the CES, once detached, could come back on Earth in less than 30 minutes, at least, some redundancy oxygen tanks can be placed in the detachable part. If properly sealed, the structure of the CES could guarantee that the escape system can maintain its nominal pressure, without expecting noticeable leakages. Under these hypotheses, the ECLSS could be placed in the rear part of the vehicle and individual oxygen systems as commercially available in aviation, could be installed within the cockpit and passengers compartment, probably accommodated under each seat, as a redundancy. This alternative could be a good compromise between risk and complexity.

Considering the environment in which the CES will operate, the early stage of development of this kind of transportation systems and the associated System Readiness Level, the use of light suits is suggested. They are not so bulky but they can provide an additional redundancy level and they can also be directly connected through proper umbilical to centralized ECLSS control and fluid distribution unit. In future, when these vehicles will have accumulated a certain amount of flight hours, the use of suits could also be avoided.

Furthermore, considering the short duration of these missions, no galleys for food have been envisaged. In case of touristic flights, the idea of the stakeholders was to equip each participant with a proper bag containing gifts and snacks.

7.9.2.7 Further considerations

In the previous subsections, the most crucial and impacting subsystems (from the point of view of mass, volume and power standpoints) have been considered. However, other items should be considered to be installed within the passengers' compartment. In particular, the overall avionic subsystem should be properly hosted in the CES. In particular, this equipment could be hosted in the cockpit and in the nose bay. In this context, special attention should be devoted to the selection and installation of antennas. This is another peculiar case in which the major requirements for the design activities strictly depend on the System of Systems in which the entire transportation system is going to be operative. In fact, the avionics and communication equipment should be selected considering the flight procedures that will be ad-hoc generated by the regulatory authorities envisaging an incremental path from the operations in restricted areas, to the integration within the civil aeronautical and space domains (Jakhu, 2011)

Eventually, special attention should be devoted to the location of the Emergency Locator Transmitter, Cockpit Voice Recorder and Flight Data Recorder within the CES. In parallel, a safety assessment should be carried out following the methodology proposed in the previous section in order to verify the level of redundancies or the system architecture aimed at maximizing the reliability of CES.

7.10 How a MBSE approach may help in the integration process.

The integration activity is one of the most complex, especially in cases where a high number of complex technologies should be putted together. In addition, the integration activity is the one of the most multidisciplinary design phases and mainly for these reasons, the overall aircraft design activity may benefit from a MBSE approach.

The reader can imagine that all the previous design steps have been carried out following a MBSE approach meaning that there is a consistent number of models with a complete traceability of the requirements onto the different design elements. The integration approach may continue on the same way simply defining the interfaces among the different models and continuing the creation of new models for the new subsystems. It is in this phase that a part from the already presented functional perspective, it would be convenient to look at the system from a behavioral and constructional perspective in order to have all the elements to move to a more complex physical description. In this section, the example of the formalization of the analysis following a MBSE is presented focusing on a specific subsystem of the reference vehicle: the air-breathing propulsion system. Indeed, also for integration purposes, the functional view is essential but it is very important to understand the way in which the several components can be logically and physically connected in order to guarantee the fulfilment of the requirements. The starting point, should always be the functional perspective and in particular, it is necessary to identify a functions to be in-depth analyzed. To show the logical composition of a system on the basis of each constituents, an Internal Block

Diagram (IBD) may be exploited. In the Figure 277 a first sketch of the logical composition of the air-breathing engine subsystem is reported while in Figure 278 and Figure 279, the representation in terms of IBD for the function "To perform vertical take-off and landing" is derived. Depending on the selected representation strategy, different IBD may be obtained. In this case, Figure 278 shows the synthetic view in which all the components with the same characteristics are grouped together and represented by a single block that has the number of components of its type as attribute. In the other representation (Figure 279), each block is a single component with its own feature. In this view, the user can define all the attributes for all the blocks but, more important, it is possible to define connection ports and relationships among blocks, that, differently from the functional diagrams, are not pure abstract connections, but they represent real flow (of data, information or physical quantities). Connection ports may be defined and characterized for the single component of the system but it is also very important to define the ports that allow the system interfacing with the rest of the model. For instance, in the example reported, it can be seen that in order to guarantee the air-breathing to be able to perform a vertical take-off and landing, the engine, that is the core component shall be fed by the air, coming from the inlet and by the fuel, coming from the tanks, through proper feeding system. Then, depending on the modes of operation of the air-breathing system in the various mission phases, the exhaust gasses coming from the combustion chamber may be redirected to different nozzles, primary or secondary ones, through the exploitation of proper distribution lines with ducts and valves.

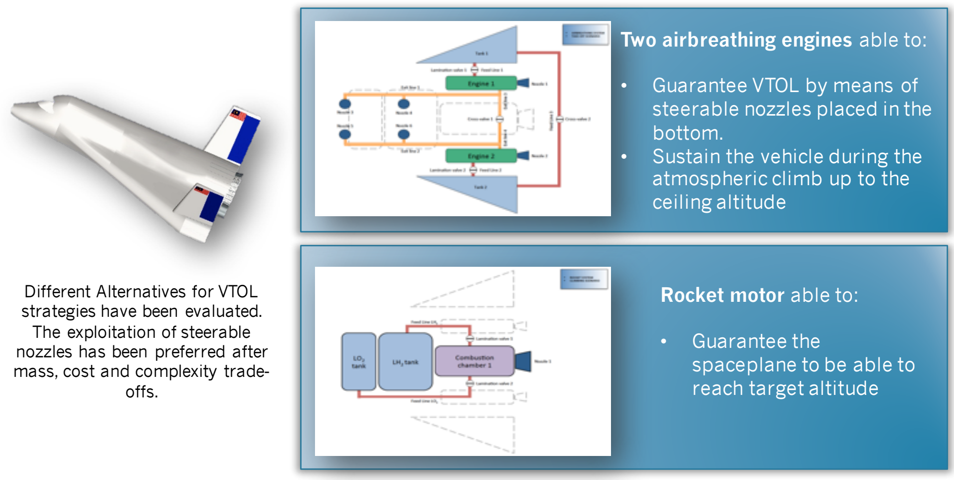


Figure 277: Summary of propulsion subsystems selected for the reference case study

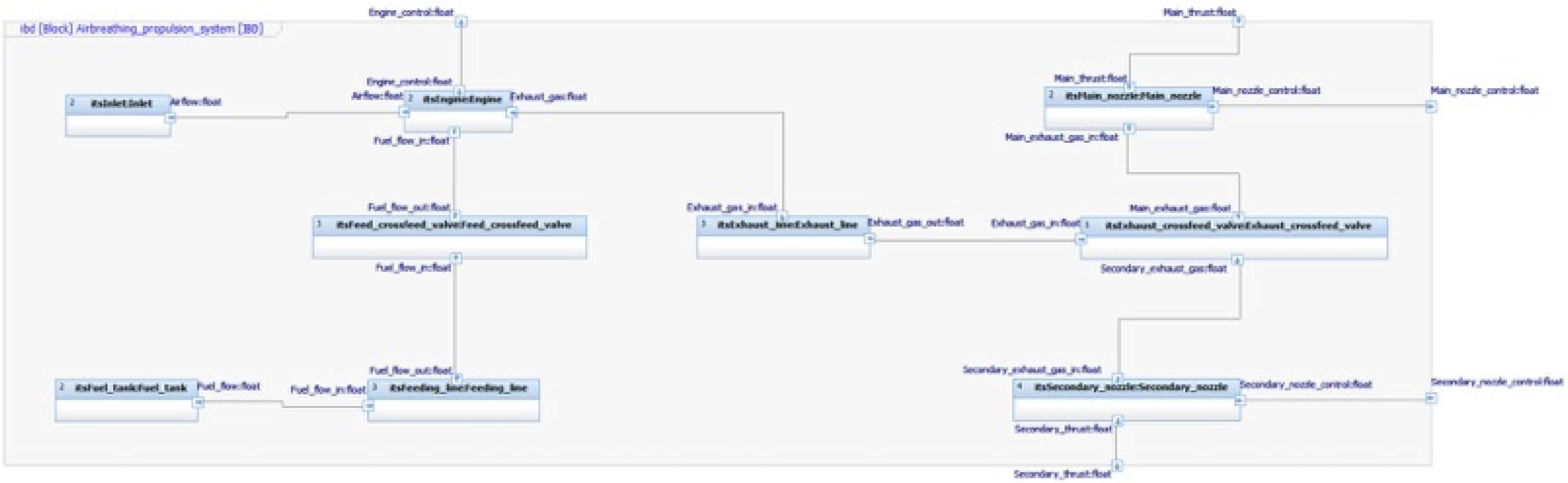


Figure 278: Synthetic IBD for the airbreathing propulsion subsystem for the reference case study

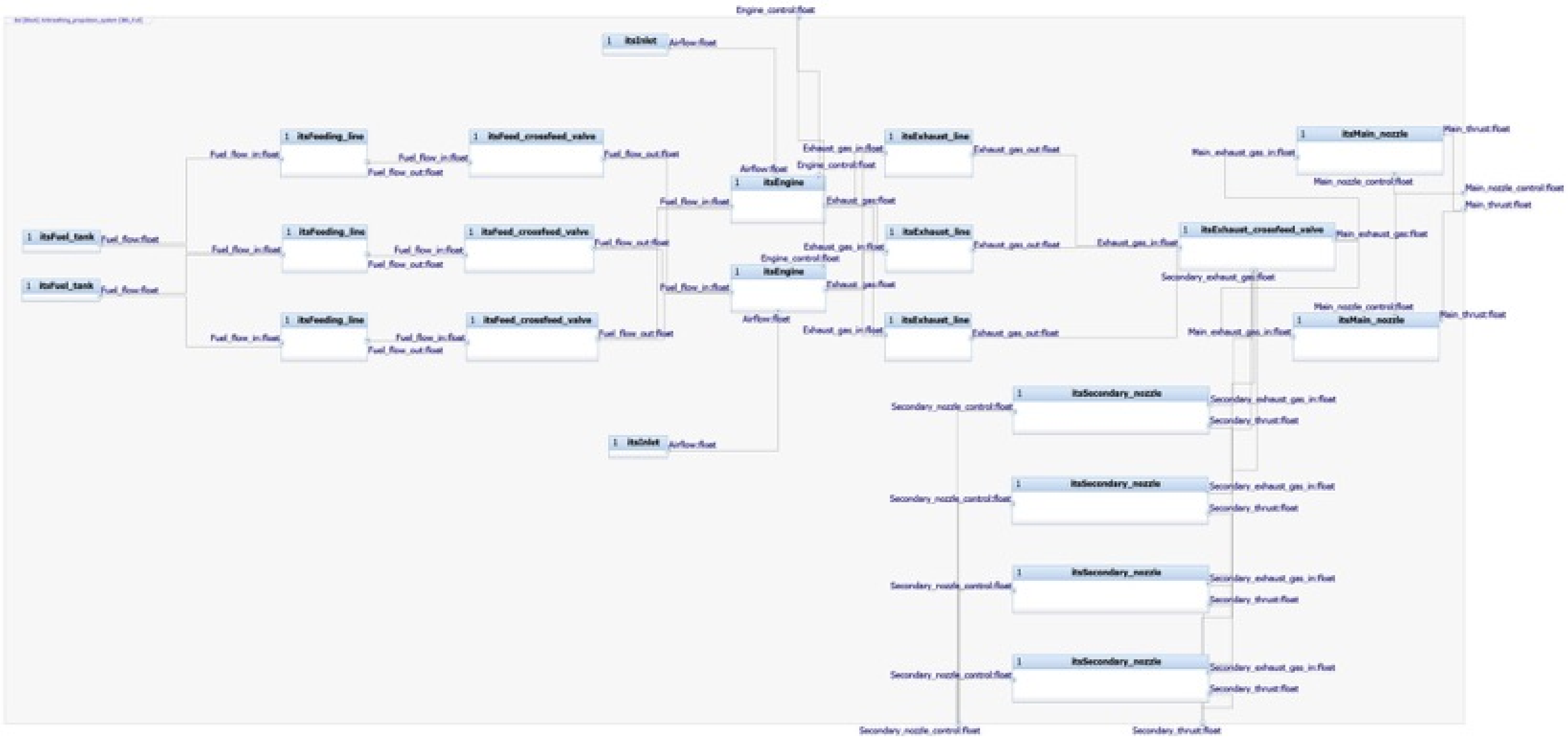


Figure 279: Extended IBD for the airbreathing propulsion subsystem for the reference case study

Instead, it is also convenient to take a look at the system from a behavioral perspective. Different diagrams may be used, each of which can help representing the system from a slightly different perspective. For this reason, the following sequence of diagrams are proposed, starting from the more generic one, moving toward those graphical representation requesting a higher level of details.

Once the function to be detailed has been selected, it is important to understand the flows of activities that should be carried out in order to achieve the reference function. For this reason, an Activity Diagram (AD) is here suggested allowing a description of the flow of actions that should be performed. The formalization is very similar to a traditional flowchart in which the several actions are related expressing transitions and messages from or to the outbox environment. In addition, in order to allow the system to move from one action to the following, some conditions must be fulfilled. Analyzing the example of the air-breathing subsystem, the Figure 280 represents a possible flow of actions that should be performed in order to allow the system to perform a vertical take-off. Once the propulsion system is activated, this can be done sending an input message to the subsystem, the main nozzle exits should be blocked preventing the horizontal thrust components to be generated, guaranteeing a pure vertical translational flight. If the thrust is higher than the aircraft weight a positive rate of climb in pure vertical translational motion is guaranteed and this condition must continue up to a certain altitude after which the thrust should be redirected towards the main nozzles allowing to move from a pure vertical translational motion, to a climb phase in which a consistent horizontal component will be present. This will be achieved through a series of actions allowing the gradual reduction of the vertical thrust up to reaching the condition in which all the exhausts are expelled through the main nozzles and thus, it is possible to switch from a vertical take-off phase to a climb mode.

In order to move to a more concrete view, it is important to move from actions and activities, to their concrete realization, i.e. to the operations. To this purpose, Sequence Diagram (SD) may be exploited in order to represent the logical time sequence of operations the system shall perform in order to satisfy a certain function. In this diagram, the main elements are the functions to be performed represented as a vertical slashed bar, that represent the proper lifeline, Additional vertical lines represent the systems borders and the actors that will be part of the operations. This formalism allows representing the operations to be performed to achieve a specific function indicating whether external inputs coming from the actors or from the environment shall be waited for or whether the operations output may be conveyed out of the system or may give feedbacks to the actors. In addition, this diagram allows to organize the operations in the proper logical sequence, allowing to define whether they are performed in parallel or in series, with respect to each other. In the example reported in Figure 281, some warnings shall be communicated back to the pilot, but it is also clear that all the operations allowing a vertical take-off may be started only if a message of propulsion switch on is communicated to the air-breathing subsystem.

The same operations can also be seen from a slightly different perspective, analyzing the sequence of modes of operations the system shall face with, to carry out a specific function. At this purpose, a State Machine Diagram (SMD), has been suggested. It can allow to model the behavior of the subsystem during its lifetime, modelling the order in which actions and activities occur and the conditions under which they occur. SMDs consist of two basic elements: states and transitions, both allowing the description of the behavior of a block over logical time. In particular, each state shows what is happening to the system or to the subsystem at any particular point in time when an object typed by the block is active. Figure 282 represents the SMD aims at describing the way of working of the air-breathing subsystems, during the take-off phase. Two major states exist: “OFF” and “ON” but within the active status, different sub-states may be activated, depending on the conditions that could happen during the mission. As well as the other presented diagrams, SMD allows a static representation of the system. However, exploiting it, it is also possible to simulate the way of working of the system under-analysis. To do this, a control panel like the one represented in Figure 283 may be designed, with the purpose of starting and control the simulation. In this case, two input parameters are present: the first is the pushbutton representing a sort of input coming from the pilot and allowing to switch on and off the propulsion subsystem. In addition, a proper control of the thrust level is present. Then, a series of customized indicators have been inserted in order to allow the designer to follow the status of the simulation. This simple simulation allows to verify whether the logical assessment of the system is able to carry out the desired functionalities on the basis of the defined input set. The exploitation of proper tool to implement a MBSE approach can also allow a live representation of the SMD, highlighting the state (block) of the system in which the system is operating in each specific point of the simulation.

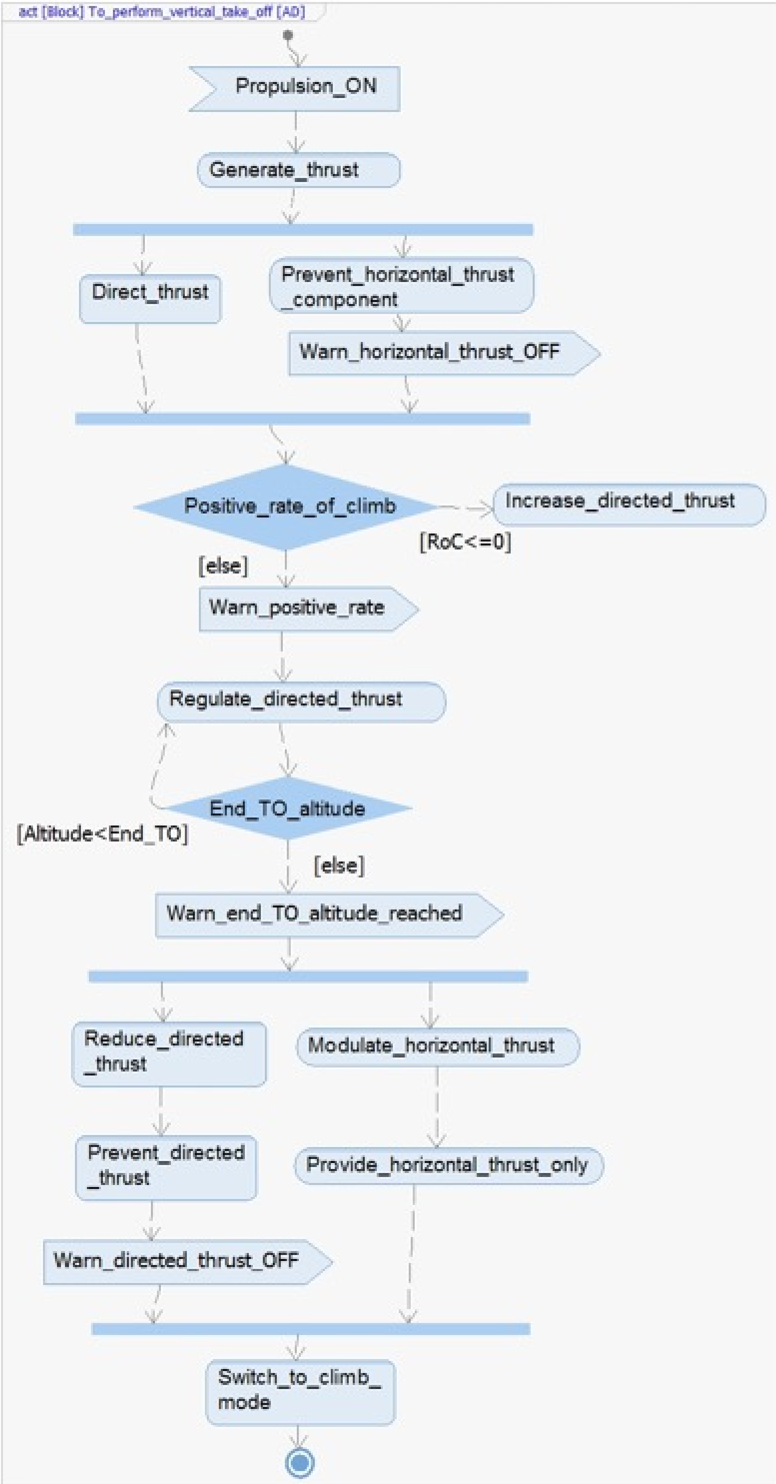


Figure 280: Activity Diagram example for the reference case study

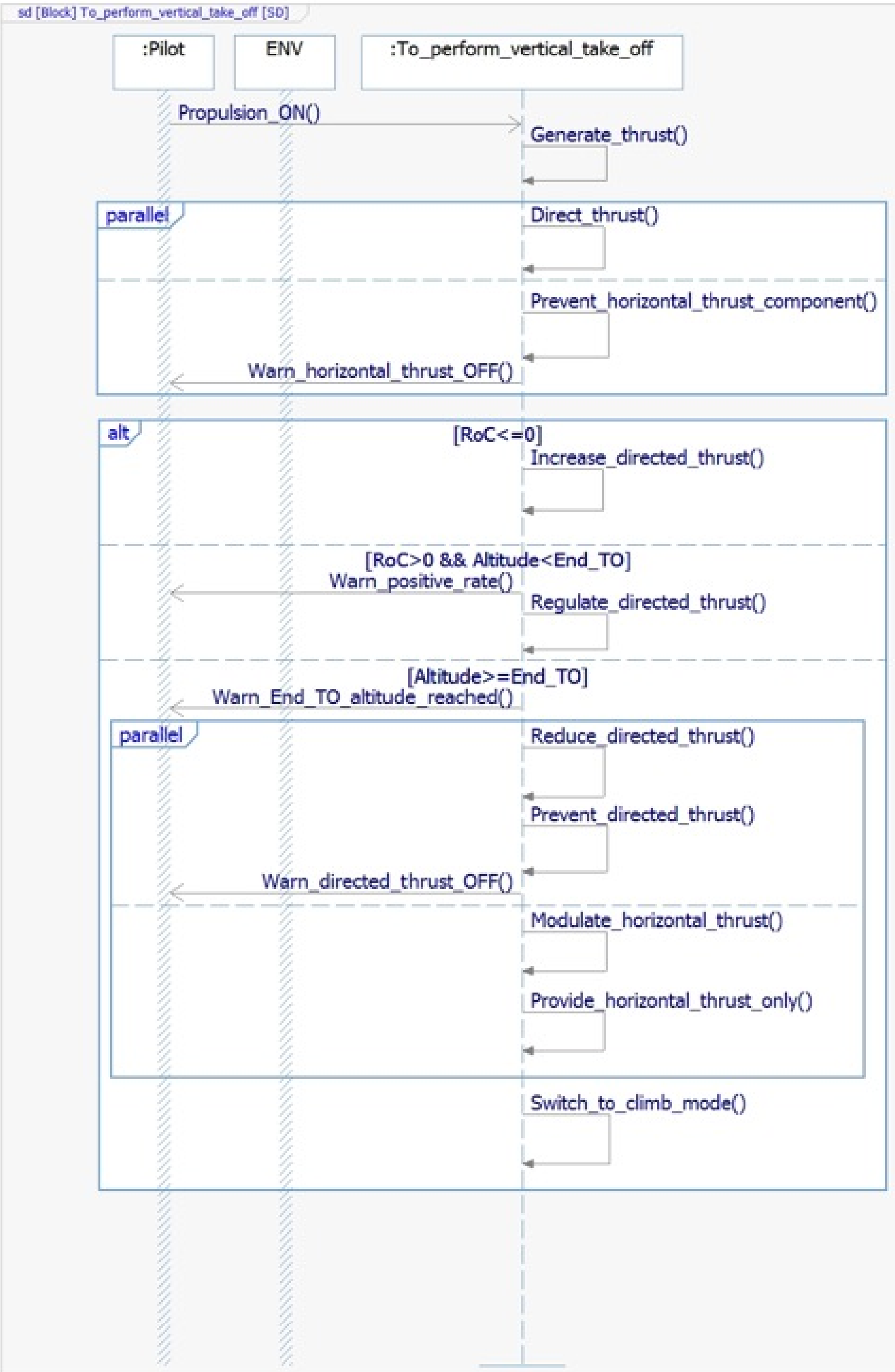


Figure 281: Sequence Diagram example for the reference case study

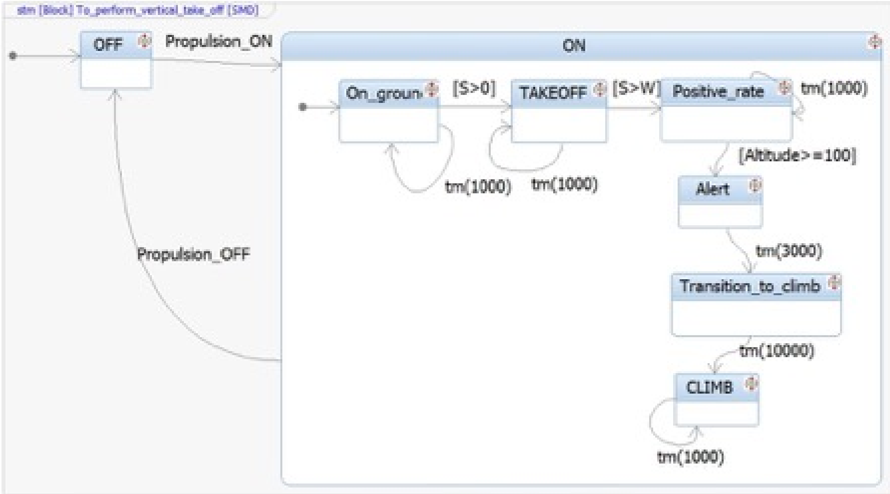


Figure 282: State Machine Diagram example for the reference case study

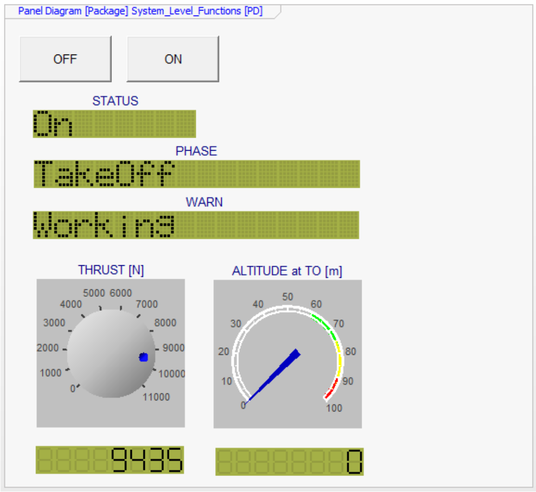


Figure 283: Panel Diagram example for the reference case study

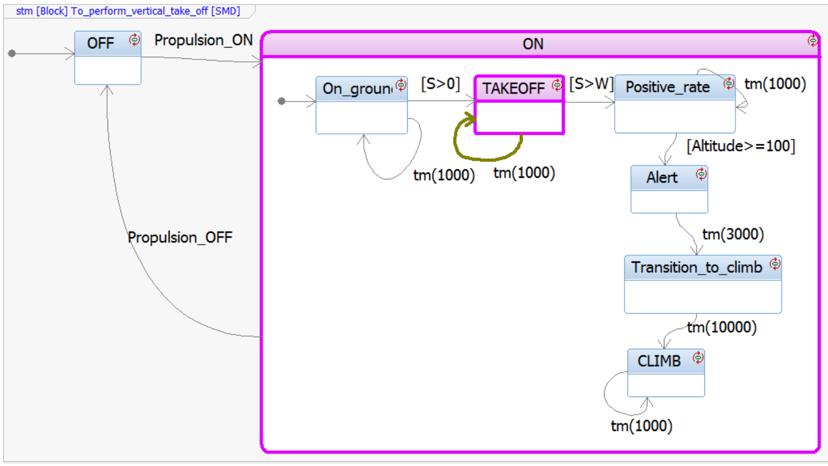


Figure 284: State Machine Diagram during a simulation for the reference case study

At this stage, it is possible to move from these views to a physical one and to this purpose, the exploitation of tools allowing CAD modelling will be exploited. It is important to notice that for the design purposes typical of a conceptual design stage, a parametric approach should be followed. In his way, it would be possible to easily implement design variations all along the design process. However, in this context the author will suggest how to exploit CAD model for different purposes.

The first is the generation of the same physical model in a dynamic simulation environment such as Simulink, Multibody SimScape Environment. In order to exploit the existing and under development links allowing tools interoperability, Solidworks® and Matlab-Simulink ® environments have been selected. Indeed, once an assembly has been fully designed in Solidworks, it is possible to export it in a Simulink environment. In particular, the several parts of the assembly become basic mechanical blocks of the SimScape library and the different constraints and relationships that have been fixed in the CAD model are here reported with the explication of the mechanical link type that would enable the envisaged behavior. In addition, all the features that are associated to the CAD model, such as the materials, masses, weight distributions directly become variables in the Simulation model. In addition, with the development of a proper Matlab code, there would be the possibility of directly modify the CAD model parameters, on the basis of some sizing refinements and to have a direct feedback on the derived and connected models. Furthermore, Simulink® allows to connect each single block and also the different variable to the requirements database, guaranteeing a complete external traceability. In this context, continuing with the example of the vertical take-off but taking a closer look to the landing gear system rather that to the propulsion one, Figure 285 shows a rapid CAD prototype developed in Solidworks® while Figure 286 aims at representing the physical scheme as it is exported in SimScape environment. This is a very simple example that allow that in later design stages could become more complex as highlighted in Figure 287 and the simulation model can also include interface with other subsystems such as the avionics allowing control of the landing gear actuations. Eventually, Figure 287 shows the interfaces allowing the requirements allocation to the different components and some example of traceability that could be reached.

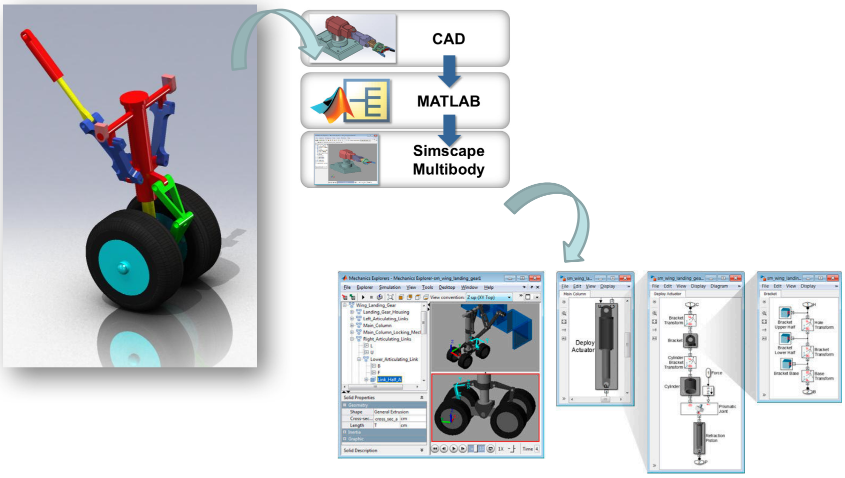


Figure 285: From CAD to Multibody simulation in Simulink

Figure 286: Example of exchange files

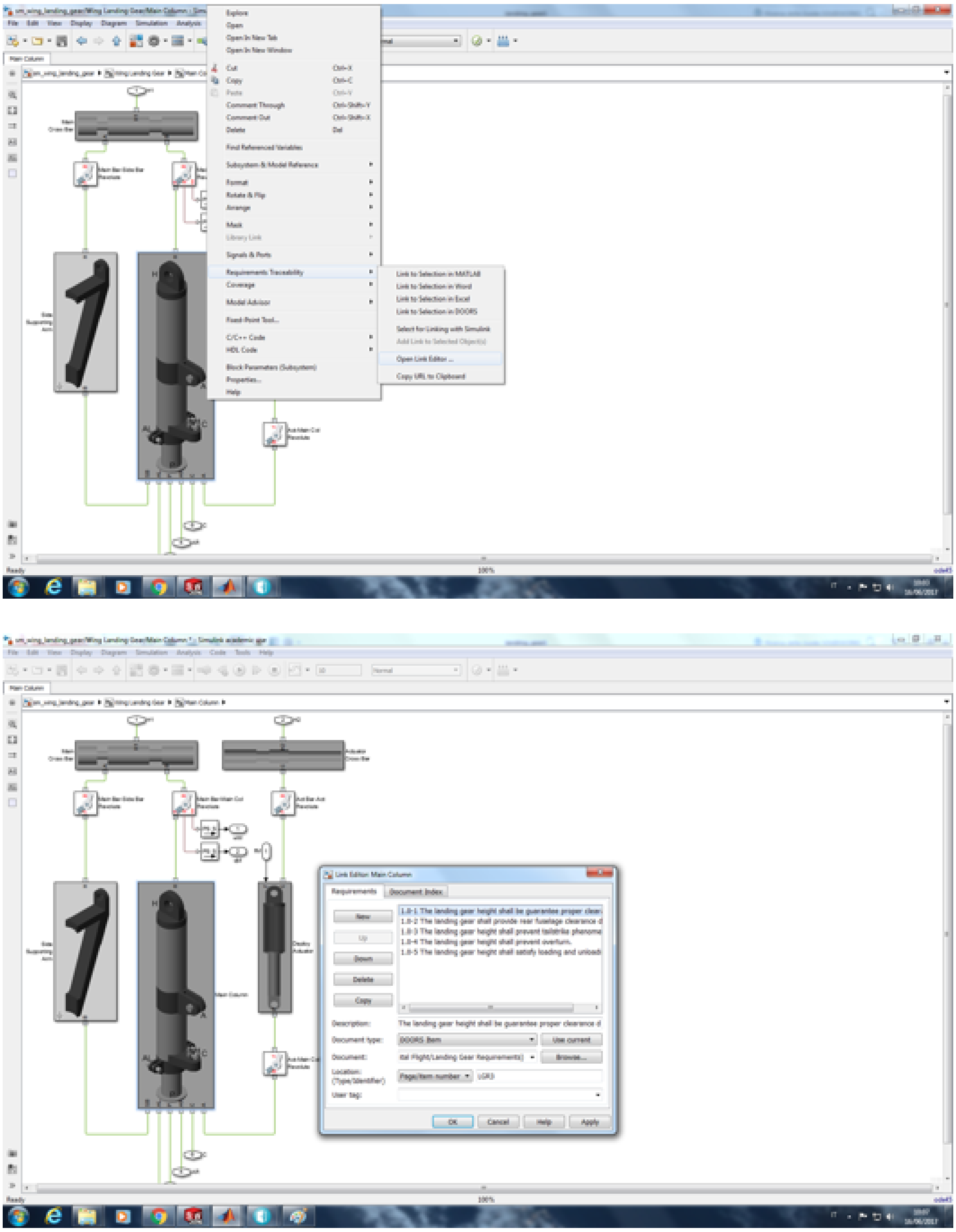


Figure 287: Requirements allocation on Simulink model

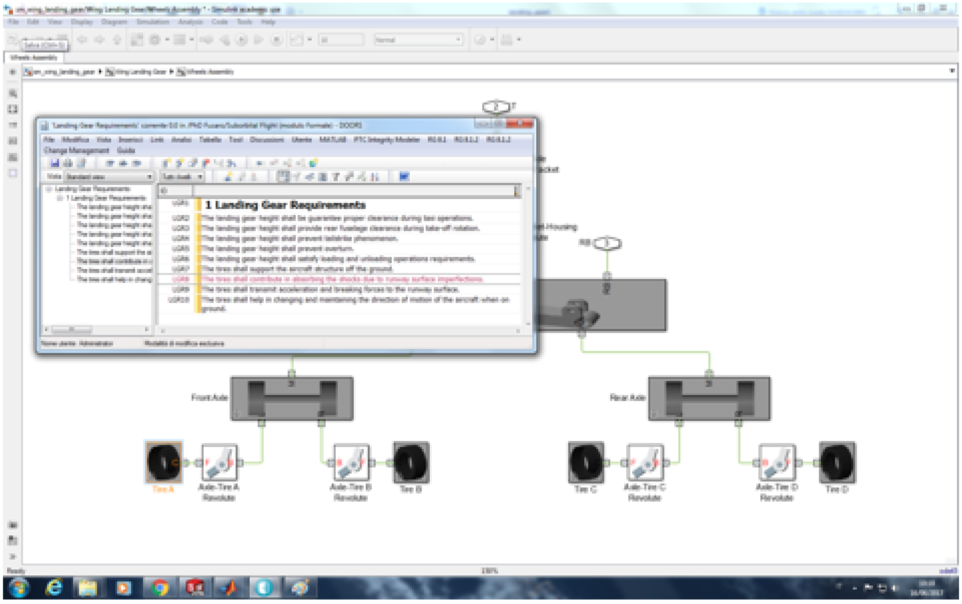


Figure 288: Example of requirements traceability

Furthermore, the generation of a CAD model will allow connections with several other tools, that may allow the possibility of generating a complex tool chain in which the innovative principle of interoperability can be guaranteed allowing to carry out a multidisciplinary design in a formalized and structured way. CAD models would be for example loaded within mission simulation environments such as STK® (see Figure 289) or as a base for additional aero-thermo-dynamic investigations (see Figure 290) or even more to start developing small scale prototype (Figure 291 and Figure 292), exploiting the rapid prototyping offered by the ultimate additive manufacturing technologies.

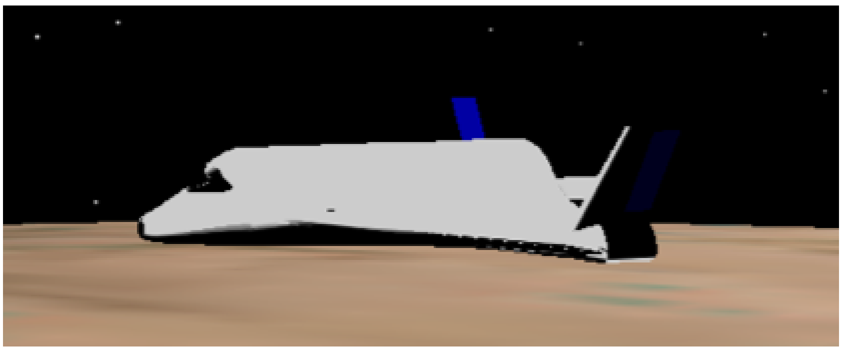


Figure 289: CAD model imported in STK® simulation environment

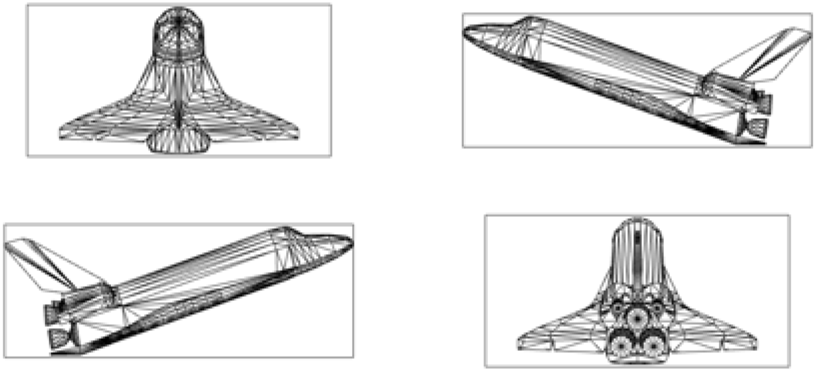


Figure 290: CAD model processed for aerothermodynamic investigations

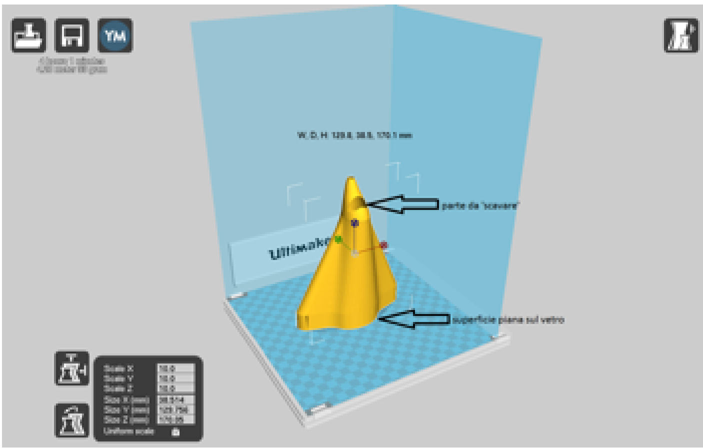


Figure 291: Example of CAD model import in software for rapid prototyping.

Figure 292: 3D printed scaled model of the reference case study.

References

Chiesa, S. (1990). Affidabilità, sicurezza e manutenzione nel progetto dei sistemi. CLUT, 1990.

De Vita, Francesco, et al. (2015) Assessment of Hypersonic Flights Operation Scenarios: analysis of launch and reentry trajectories, and derived top level vehicle system and support infrastructure concepts and requirements. *20th AIAA International Space Planes and Hypersonic Systems and Technologies Conference*., Glasgow (UK).

Jakhu, R. S., et al. (2011). The Need for an Integrated Regulatory Regime for Aviation and Space: ICAO for Space?. Vol. 7. *Springer Science & Business Media*.

Musgrave, G. E., et el. (2009). Safety design for space systems. Butterworth-Heinemann.

Peffer, Andrew, et al. (2000) Development and transition of low-shock spacecraft release devices. *Aerospace Conference Proceedings, 2000 IEEE*. Vol. 4. IEEE.

Raymer, D. P., (2012). Aircraft design: A conceptual approach, *American Institute of Aeronautics and Astronautics*. Inc., Reston, VA (2012): 21.

SAE (1996), ARP 4761, Guidelines and methods for conducting the safety assessment process on civil airborne systems and equipment, Warrendale (PA).

Tumino G. at al. (2008) The IXV Project. The ESA re-entry system and technologies demonstrator paving the way to European autonomous space transportation and exploration endeavours. *IAC-Paper IAC-08-D2.6.1*

Victor M., et al. (2016) The IXV guidance, navigation and control subsystem: Development, verification and performances. *Acta Astronautica* 124: 53-66.

Villace, V. F., et al. (2015) The Thermal Paradox of Hypersonic Cruisers. *20th AIAA International Space Planes and Hypersonic Systems and Technologies Conference.* 2015.

Conclusions

This Thesis aims at suggesting an innovative integrated and multidisciplinary design methodology for conceptual and preliminary design phases with the goal of allowing designers to the realization of innovative aerospace products to be more competitive on the market, more environmentally friendly and safe.

In addition to the theoretical description of the methodology, the Thesis suggested a proper tool-chain based on a Model Based Systems Engineering approach allowing the integration in a unique work-flow of all the activities envisaged for the conceptual and preliminary design phases. Indeed, the methodology, as well as the envisaged tool chain are able to support different activities, from the stakeholders' analysis, to generation of the strategic decision plans, from the mission scenario alternatives generation to the selection of the optimal baseline, from the qualitative aircraft layout definition to the high level quantitative estimations, from the design of the most impacting systems to their integration on board. During all this process that starting from the identification of possible private and public interests lead to the identification of the several technologies required to satisfy their expectations, a crucial role is played by requirements. Requirements are elicited all along the design process but are also collected and organized in proper databases enhancing the traceability and allowing their allocation at different levels, from systems to basic components. Another important role is played by trade-off analyses; indeed, all along the Thesis several trade-off examples have been presented and in all the situations, specific algorithms able to move from qualitative-based to quantitative-based decisions have been suggested with many advantages also in terms of replicability of the design process and traceability of the selection processes. Moreover, all along this document, safety considerations have been specially taken into account, since mission level, up to the design of proper systems solutions to diminish the risk of loss of lives in aeronautical and space missions,

For the sake of clarity and to strength the importance of such design methodology, both the suggested approach as well as the envisaged tool chain has been applied to a specific reference case study. This implementation example is inspired by a real pre-feasibility study aiming at designing a suborbital transportation system able to perform suborbital parabolic flight services enhancing to few passengers at a time to experience an astronaut-like experience. In addition, to enlarge the possible locations from which these missions would be performed, the vehicle should be able to perform vertical take-off and landing without exploiting rocket motors up to a certain altitude. Through a rigorous application of the suggested integrated and multidisciplinary design methodology, a proper mission and vehicle layout have been obtained. Furthermore, the exploitation of the envisaged tool-chain has guaranteed several benefits to the overall design process: the reduction in development time also possible thanks to the even increasing interoperability among selected tools, the complete traceability of requirements to each single design element, the possibility of ease the iterative and recursive process. Moreover, technological advancements have been suggested such as the introduction of O-LED panels to avoid structural discontinuities and related drawbacks or the design of an integrated Cabin Escape Systems able to be detached from the mother-spacecraft in case of catastrophic events.

In addition, this Thesis provides useful suggestions to carry out a mission and vehicle design for the future evolution of commercial suborbital vehicles, i.e. hypersonic and re-entry vehicles. For this reason, all along the document, special attention has been devoted to the selection of reference values for the hypersonic case or to the identification of the major challenges the designer should face with in case of the design of a transportation system aimed at flying at hypersonic speed.

Eventually the present work only paves the way for several other research activities that could benefits from the results obtained by these doctoral studies; in particular, the integration of a more in-depth costs analysis may be envisaged as well as the application of the overall methodology with relative tool chain, to the design of a different case study, as for example a hypersonic point-to-point transportation system.

Last but not least, the suggested work-flow with the envisaged tool-chain has been developed thinking not only to a rationalization of the current industrial design process, but also to support high-level education of the current and future generations of aerospace design engineers.